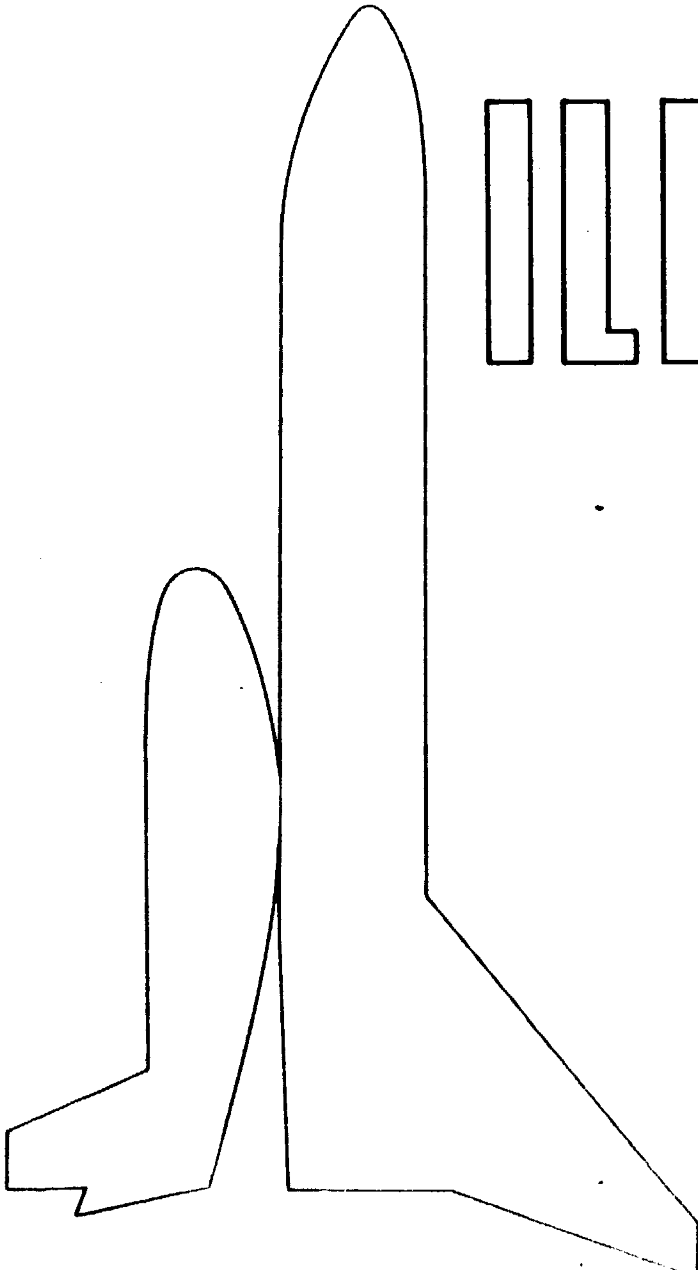


ILRVS

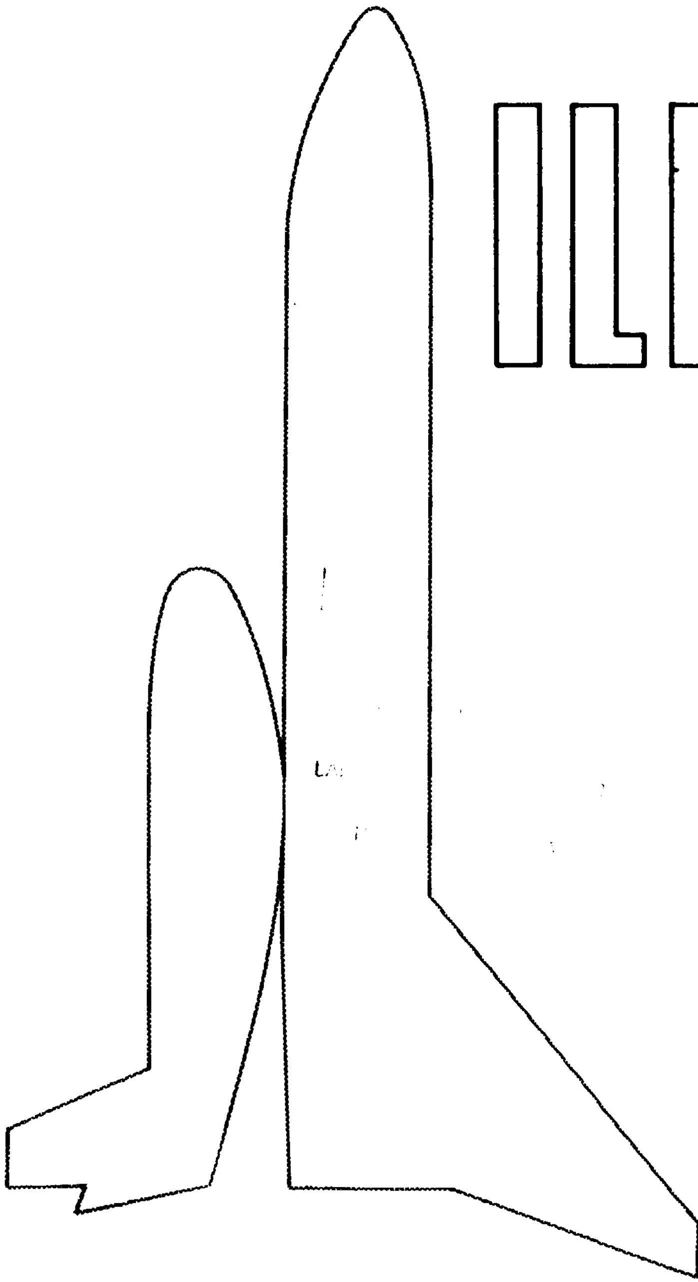
Integral Launch and Reentry Vehicle System

Final Oral Presentation

4 November 1969

Contract NAS 9-9204

MCDONNELL DOUGLAS CORPORATION



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Integral Launch and Reentry Vehicle System

Final Oral Presentation

Report MDC E0039
4 November 1969
Contract NAS 9-9204

MCDONNELL DOUGLAS
CORPORATION

INTRODUCTION

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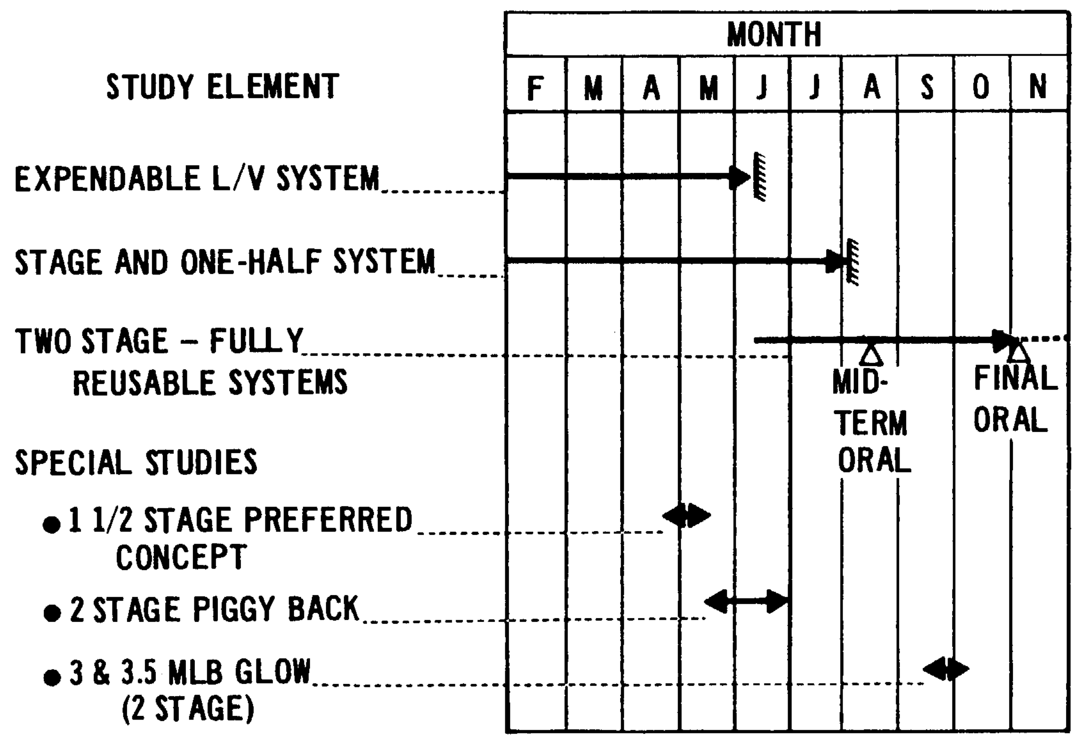
ILRVS STUDY EVOLUTION

The study emphasis has shifted drastically during the past 9 months. Final reports have been submitted for the Expendable L/V System portion. Because of the discontinuation of the expendable and stage-and-one-half efforts, the final oral presentation is limited for the remaining concept of interest - two-stage fully reusable.

Several special efforts during the study period have materially reduced the desired depth of analysis in the mainline effort. However, these special activities were not without benefit in that design approaches and critical sensitivities were surfaced that might have otherwise gone unexamined.

ILRVS STUDY EVOLUTION



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CONCEPT EVOLUTION

Significant changes in the baseline concept have occured during the study period. The adjoining chart summarizes some of the major increments. In all but two cases the change has resulted in a significant improvement in system performance. The 10% contingency item was made to conform to study ground rules. The cruise back to launch site was selected for operational advantage after the sum of other changes resulted in a first stage cruise back range on the same order as that required to recover at a remote site.

The chart demonstrates a significant learning process during the course of the study.

CONCEPT EVOLUTION

ITEM	MID-TERM	FINAL
BOOST MODE	10% PUMPED IDLE	SERIES BURN
HL-10 TANKS	NON-INTEGRAL	INTEGRAL
INERT WEIGHT CONTINGENCY	0	10%
CARRIER RECOVERY	REMOTE SITE	LAUNCH SITE
AERO SURFACE STRUCTURE	COLD	WARM
CARRIER PROPELLANT VOLUME	REFERENCE	1.09 REFERENCE
CARRIER WING	HIGH	LOW

ILRVS-446F

SPECIAL STUDY UPDATE

On 2 October 1969 a special report was submitted to the Manned Space Flight Council at the direction of Langley Research Center. The effort encompassed the sizing and design of two fully recoverable two-stage systems with ground rules that differed significantly from the mainline effort. Because of the short reaction time allowed, it was impossible to inspect all potential trades that affected the system's payload capability - not to mention the simple mechanical errors inherent in panic efforts. Consequently, the data must be considered "soft", as must any conclusions drawn therefrom.

In order to support the above contention, the data on the facing page shows how a little more time for analysis - this work was done subsequent to the report submission - can drastically alter the conclusions. At the time of report submission one could only conclude that the 3.5MLB configuration was marginal and the 3.0MLB configuration was unacceptable. Furthermore, the large ballast penalties quoted suggests the system is fundamentally incompatible with the requirements.

The current data shows the 3.5MLB configuration to have excellent payload capability and the 3.0MLB case to be marginal. In both cases the ballast has essentially disappeared thus countering the incompatibility argument.

With even more work, equally dramatic changes may well be forthcoming.

SPECIAL STUDY UPDATE
(MSF Management Council)

HL-10 ITEM	10/2/69 REPORTED WEIGHT (LB)	INCREMENT	
		WEIGHT (LB)	BALLAST (LB)
VERTICAL TAIL	6,460	–1,250	– 900
SIDE FINS	12,240	–1,450	–1,050
ELEVONS	12,710	–6,650	–5,000
THRUST STRUCTURE	4,100	–1,230	– 900
BODY TPS	24,610	0	–2,000
TOTAL	–	–10,580	–9,850
BALLAST	10,000	– 9,850	
CARGO			
3.5 MLB CONFIG	24,000	+20,430	
3.0 MLB CONFIG	0	+20,430	

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PRESENTATION OUTLINE

The elements of the presentation are composed of the seven special emphasis areas (items 5 through 11) and the basic study tasks (items 3, 4, 12 and 13). Two tasks are not covered independently - Mission Analysis and Program Plans. The former became a sub-element of item 10 (Mission Interfaces and Cargo Handling), and the information content of the latter is redundant to the information available throughout the other topic areas.

Presentation of topics 1 through 8 and 13 was given by H. C. Vetter, Study Manager; and topics 9 through 13 was presented by D. L. Sturgis, Principal Analyst.

PRESENTATION OUTLINE

1. INTRODUCTION
2. BASELINE SYSTEM SUMMARY
3. SYSTEM DESIGN & OPERATIONAL SENSITIVITIES
4. CONCEPTUAL DESIGN DEFINITION
5. REENTRY HEATING AND THERMAL PROTECTION
6. PROPULSION SYSTEM PARAMETERS
7. ABORT
8. APPROACH AND TERMINAL LANDING
9. INTEGRATED ELECTRONICS
10. MISSION INTERFACES AND CARGO HANDLING
11. GROUND TURN-AROUND OPERATIONS
12. SCHEDULES AND COSTS
13. TECHNOLOGY IDENTIFICATION
14. SUMMARY AND RECOMMENDATIONS

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BASELINE SYSTEM SUMMARY

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BASELINE MISSION PROFILE (2ND STAGE)

The carrier-orbiter system lifts off and rises vertically for 20 seconds. It is then pitched slightly away from vertical so gravity turn that follows takes it to the desired separation conditions. At separation the carrier pitches to 50 degrees angle of attack while inverted (bank angle = 180°).

Inverted flight at high angle of attack continues through apogee to a flight path angle three degrees below horizontal. This is for the purpose of losing velocity and "digging-into" the atmosphere so that cruise back range is minimized. A roll-out to unbanked flight is then performed to allow a pullout well-above thermodynamic boundaries. Pullout is defined as a flight path two degrees below horizontal.

After pullout the carrier is banked 70 degrees while maintaining 50 degrees angle of attack. This attitude is selected as producing the lowest practical lift-drag ratio and hence the shortest practical range. A steeper bank would shorten the range slightly, however, there would be a significant penalty in the severity of the environment.

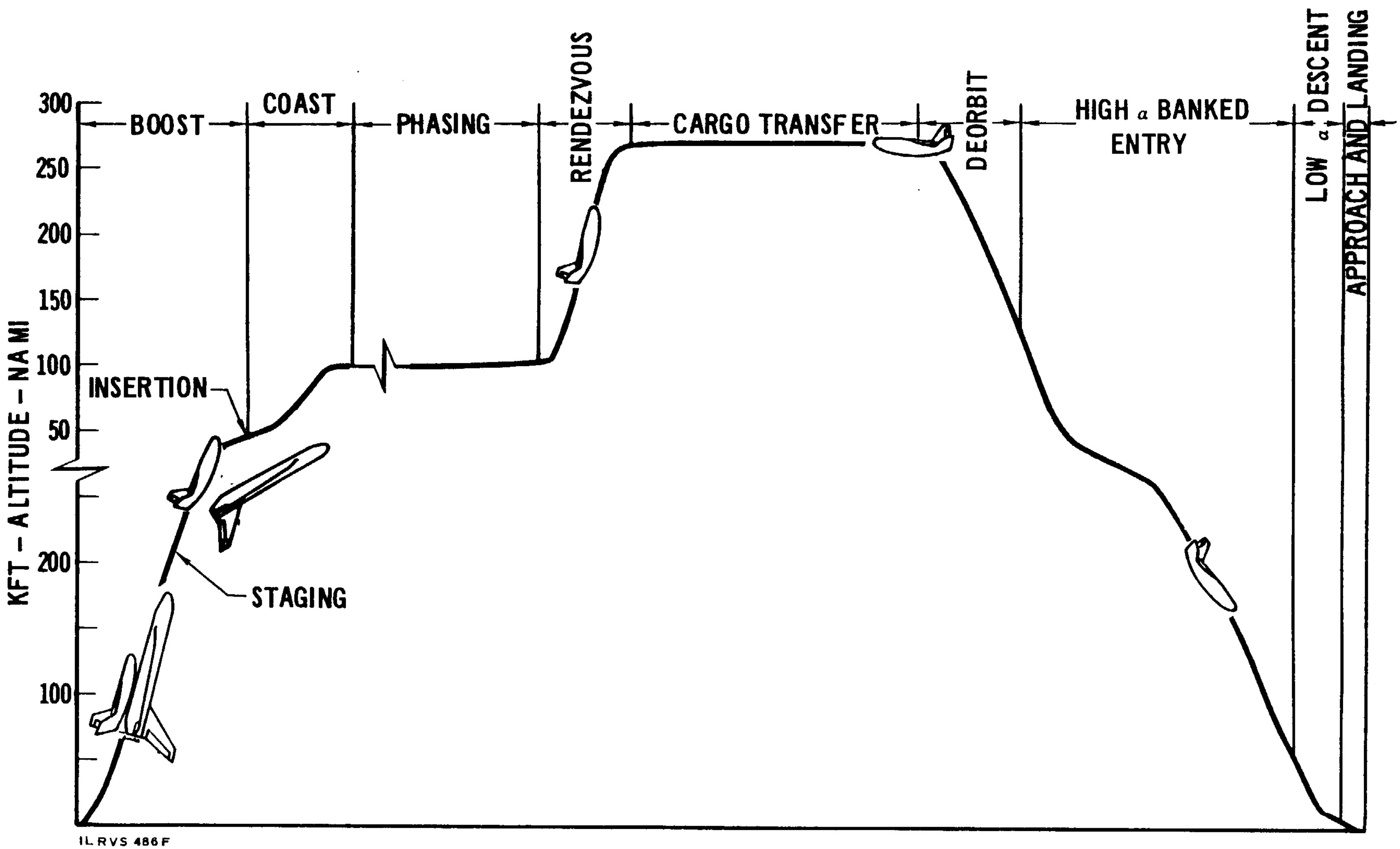
By a speed of about 1000 ft/sec the carrier has been turned to face the launch site. It is then rolled to a wings level attitude to begin the transition to cruise conditions. Angle of attack is modulated to produce a constant load factor which yields horizontal flight at 10000 feet.

When horizontal flight is reached the engines are deployed and the carrier decelerates to mach .36. Cruise back to the launch site then follows at constant altitude and speed.

The landing maneuvers is then initiated by . . .

BASELINE MISSION PROFILE

Second Stage



BASELINE MISSION PROFILE (1ST STAGE)

The HL-10 is lifted off in a belly-to-belly configuration with the carrier. The carrier takes it to a separation velocity of about 9000 ft/sec, where HL-10 engines are ignited.

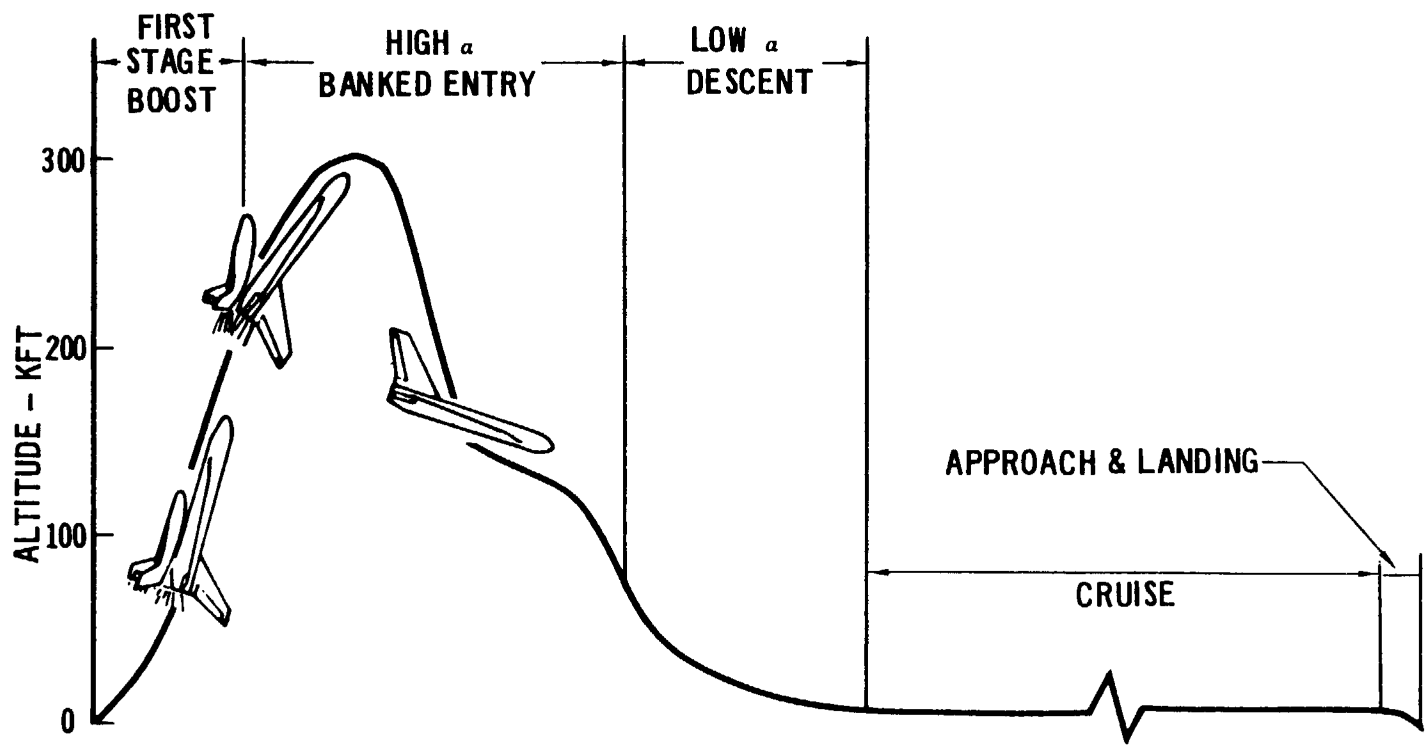
Thrust deflection is modulated to attain the desired orbit insertion conditions while thrust magnitude is modulated so as to not violate the axial load factor constraint. Orbit insertion is at perigee of a 45 X 100 nautical mile orbit.

After a coast to apogee, sufficient posigrade impulse is applied to circularize at 100 nautical miles. This 100 nautical mile orbit is then maintained to get in phase for later rendezvous. At the proper time an impulse is applied to raise apogee to 270 nautical miles and then when apogee is reached another impulse makes the orbit circular. Rendezvous is achieved; cargo is transferred; and payload transfer is performed. The HL-10 then separates and with a retrograde impulse de-orbits to a reentry velocity of 25990 ft/sec and a flight path angle of -1.5 degrees at 400000 feet.

At reentry, the HL-10 is rolled 65.8 degrees and then pitched 50 degrees. This attitude is maintained to a velocity of 24500 ft/sec where a bottom centerline temperature of 2200°F results. From that point bank angle is continuously modulated to maintain 2200°F until a velocity of about 17000 ft/sec is reached. At that speed the 3g normal load factor constraint becomes more restrictive and bank angle is modulated to fly 3g's. The purpose of flying along these constraints is to minimize time and hence total heat. At about 10000 ft/sec a slow roll to wings level is begun. This is done to stretch out the range and satisfy the 390 nautical mile cross range requirement for once-a-day return.

At 3900 ft/sec the HL-10 is pitched down to a zero lift attitude and allowed to fall to 45000 feet angle of attack is modulated to pull 2.7 normal g's while bank angle is held at 50 degrees. This gives the proper lift-drag relationship to reach the high key point where the landing maneuver begins.

BASELINE MISSION PROFILE
First Stage



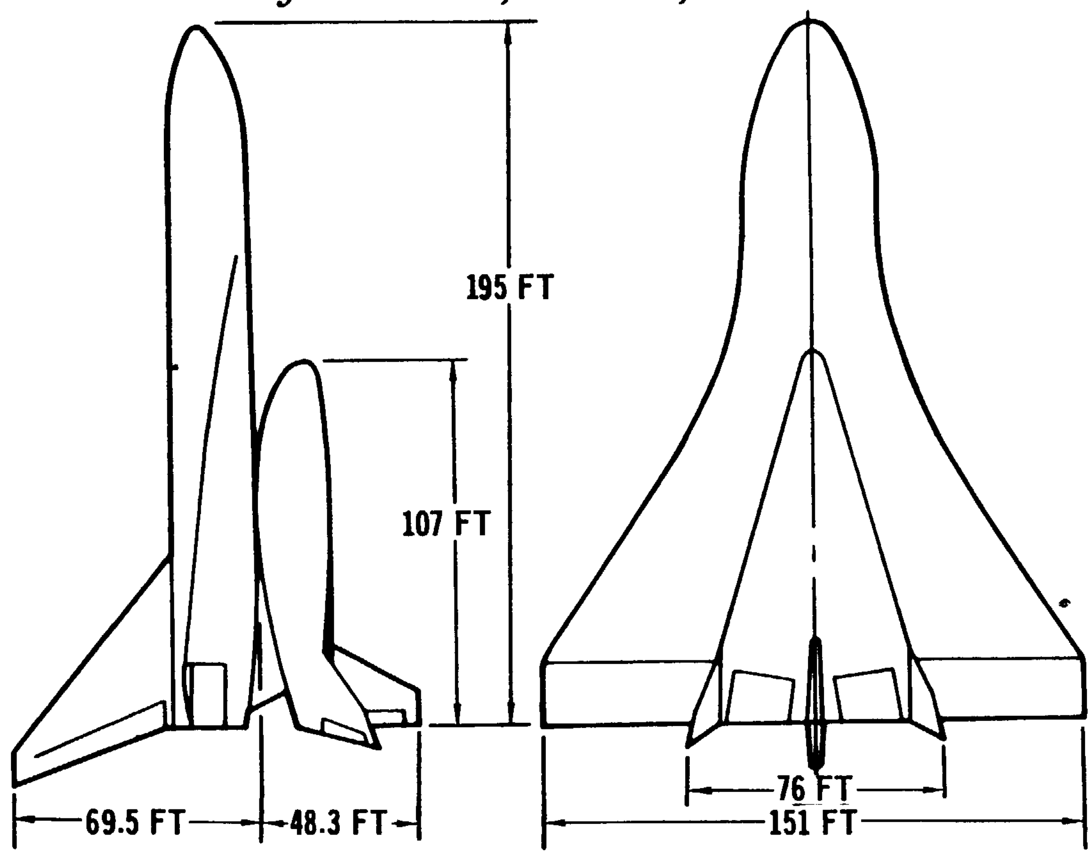
LAUNCH CONFIGURATION

The launch configuration for the 25,000 lb. cargo is shown here. The orbiting vehicle (HL-10) is sized for a 16 ft. dia., 32 ft. long cargo bay centered on the vehicle c.g. The HL-10 length required for this constraint is 107 ft. A major portion of the remainder of the HL-10 is filled with boost propellant.

The carrier vehicle, a clipped delta wing configuration, contains the remainder of the boost propellant required to achieve the required launch performance. The length of this first stage is 195 ft.

The two vehicles are oriented at launch with an interface between their lower surfaces. The bases of the two stages are in the same plane. The overall height of the launch configuration is limited to the 195 ft. carrier length. This arrangement provides a stable aerodynamic configuration throughout boost, low sensitivity to ground winds, lowest access height to cargo and orbiter, and retains the option of firing the orbiter engines while mated.

BASELINE LAUNCH CONFIGURATION
Payload -25,000 Lb,15' × 30'



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VEHICLE PERFORMANCE SUMMARY

The significant performance parameters for the two stage vehicle system are discussed below:

Impulsive Velocity - The launch ΔV split (ideal) between the first and second stage vehicles is determined primarily by the volume available in the 107 foot HL-10. The maximum ΔV that can be incorporated in the orbiter is 16,777 ft/sec so the remainder of the 31,250 fps required establishes the size of the clipped delta. On-orbit ΔV of 2000 fps is in the HL-10 for orbit maneuvering and attitude control.

Propellant I_{SP} - The bell engine used in both stages has a two position nozzle and the expansion ratios are different in each stage. Information about the expansion ratios and propellant I_{SP} values for the orbiter and carrier LO_2/LH_2 systems can be found in the classified annex to this brochure.

Propellant Fraction - Propellant fraction for the first stage is defined to be the ratio of usable launch propellant plus idle mode propellant to gross launch weight at ignition. The fraction in the second stage is defined to be the ratio of usable launch propellant to second stage weight at separation.

Maximum Acceleration - The launch trajectory for the integrated system as well as the entry trajectories for both stages were shaped so as not to exceed a 3g boundary. In the launch trajectory the system was throttled back to keep the g-limit to 3. During entry, the carrier and orbiter approached but did not reach the 3-g limit.

Cruise Range - The first stage vehicle has a cruise range of 340 NM plus approach and go-around contingency for head-winds and hot-day operation. This range is sufficient for the carrier to return to the launch site. Wi once a day return capability the HL-10 has no cruise requirement.

Wing Loading - The wing loading (W/S) for each stage is based on the total vehicle projected planform area. These areas are 13,300 ft^2 and 4,160 ft^2 for the carrier and orbiter respectively.

Touchdown Speed - The touchdown velocity for the carrier (137 knots) is based on a touchdown angle of 12°. Touchdown angle for the HL-10 is 23°.

VEHICLE PERFORMANCE SUMMARY

PARAMETER	CARRIER	ORBITER
IMPULSIVE VELOCITY		
● LAUNCH	14,473 FPS	16,777 FPS
● ORBITAL	–	2,000 FPS
MAXIMUM ACCELERATION	3 g	3 g
PROPELLANT FRACTION	0.808 (BOOST)	0.684 (BOOST)
CRUISE RANGE	340 NM PLUS APPR. & LANDING AND CONTINGENCY	APPR. & LANDING
WING LOADING (W/S)*		
● ENTRY	37 PSF	47 PSF
● LANDING	33 PSF	44 PSF
TOUCH DOWN SPEED	137 KNOTS	171 KNOTS
● T.D. α	12^{0}	23^{0}

*BASED ON TOTAL PROJECTED AREA

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TWO STAGE WEIGHT AND MISSION HISTORY SUMMARY

Abbreviated group weight statements are presented for both the 107 ft. HL-10 (second stage) and the 195 ft. carrier (first stage) vehicles. The weights represent a total launch vehicle which is capable of satisfying the study requirements.

The mission history is a normal mission from gross weight on the pad before ignition thru landing. The only removals thru those mission points is the expenditure of propellants. Dry weight is derived by removing the residual propellants, cargo and crew weights from the landing weight.

TWO STAGE WEIGHT SUMMARY
Baseline System

VEHICLE	ORBITER	CARRIER
LENGTH–FT	107	195
THERMOSTRUCTURES	91,470	249,600
LANDING SYSTEM	9,190	22,310
MAIN PROPULSION SYSTEM	527,750	2,302,860
SECONDARY PROPULSION SYSTEM	38,790	2,250
LANDING PROPULSION SYSTEM	24,710	100,900
SUBSYSTEMS AND CREW	13,310	11,070
CARGO	25,000	–
GROSS PAD WEIGHT	730,220	2,688,990
LIFTOFF WEIGHT	730,220	2,671,920
SECOND STAGE SEPARATION	730,220	–
INJECTED WEIGHT	230,229	511,880
RETROGRADE WEIGHT	197,690	511,880
ENTRY WEIGHT	195,760	511,880
LANDING WEIGHT	185,800	450,940
DRY WEIGHT	156,470	438,230
GROSS LIFTOFF WEIGHT	3,402,140	

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WEIGHT FRACTIONS SUMMARY

This chart is presented to illustrate fractionally the relationships of the weights on the previous chart (Two-Stage Weight Summary). The relationships chosen are those which are generally of interest in a mass fraction basis. Of interest is the thermal-structures fraction which is over 50% for both vehicles and the propellant mass fractions which are somewhat low in comparison to normal launch vehicles. However, this is a reflection of the first stages manned status and its ability to return and the second stages cargo carrying fraction as well as boost. The payload fraction also appears low by comparison to earlier vehicles. However, it must be remembered that this payload is the actual cargo that can be transferred to a space station and not the usual weight to orbit definition.

WEIGHT FRACTIONS SUMMARY

	ORBITER	CARRIER
THERMO-STRUCTURES	58.4% DRY	56.9% DRY
PROPULSION INERTS	27.6% DRY	35.5% DRY
PROPELLANT	74.5% GROSS	83.1% GROSS
• BOOST	68.4% GROSS	80.8% GROSS
• ORBIT (MAN & ACS)	14.9% INJECTED	0.2% BURN-OUT
• CRUISE/LANDING	6.1% ENTRY	11.9% BURN-OUT
SUBSYSTEMS	14.0% DRY	7.6% DRY
PAYLOAD	16.0% DRY 3.4% GROSS (ORBITER)	21.5%* GROSS (ORBITER + CARRIER)

*ORBITER GROSS WEIGHT

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SYSTEM DESIGN AND OPERATIONS SENSITIVITIES

Task 6

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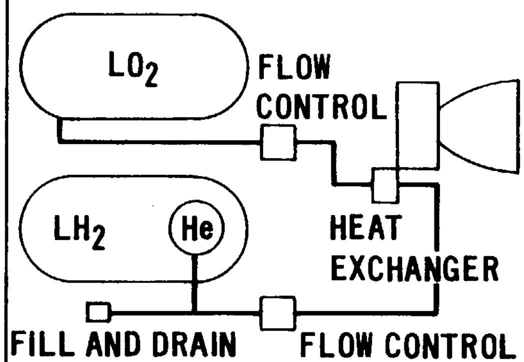
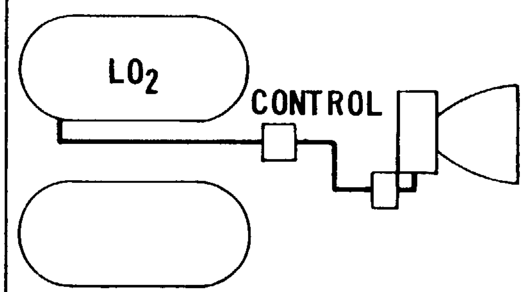
GROUND RULE SUMMARY

A summary of the major ground rules for the study is summarized on the facing page.

BOOST OXYGEN PRESSURIZATION CONCEPT

	ARRANGEMENT	WEIGHT (LB)		
			1ST STAGE	2ND STAGE
HEATED HELIUM		PRESSURANT & BOTTLES	6100	1290
		LINES & VALVES	3060	616
		TOTAL	9160	1906
			1ST STAGE	2ND STAGE
√ GASEOUS OXYGEN (GOX)		GOX	8850	1870
		LINES & VALVES	790	158
		TOTAL	9640	2028

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SECONDARY PROPULSION SYSTEM CONCEPTS

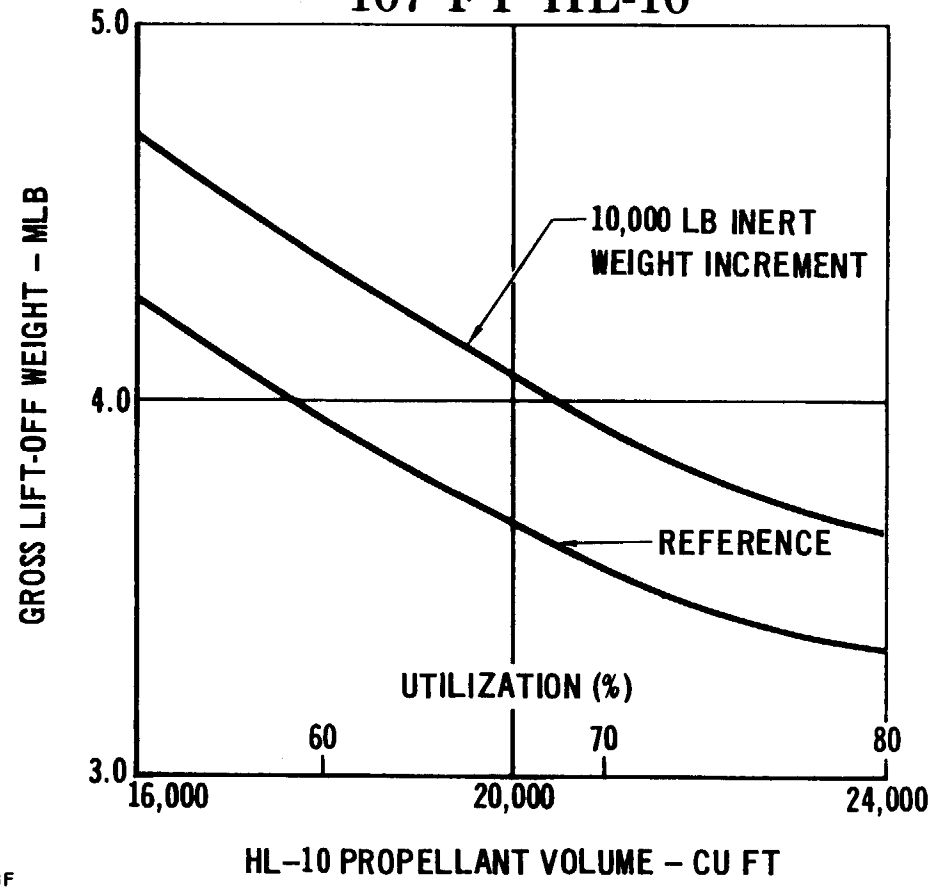
Three engine configurations were considered to perform the main orbit maneuvers of rendezvous and deorbit; a throttled boost engine, installation of separate RL-10 engines and addition of aft firing attitude control (ACS) engines. A comparison of these alternate concepts is presented on the opposite page.

The throttled (10%) boost engine, although having a high I_{sp}, is the highest weight concept due to the large amount of propellant lost during start up and shut down. No engine installation penalty,is incurred, however, additional feed lines and flow components are necessary. Engine development is effected because of increased number of starts per mission as well as the increased importance of repeatable start up and shut down transient performance. The RL-10 engine configuration which exhibits the best combination of the high I_{sp} and small propellant losses, is the minimum weight concept. However, a modification to the 2nd stage base geometry is required to provide sufficient installation space. Separate feed lines and flow control components must also be added. Further qualification of the RL-10 is needed to demonstrate the additional burn time and restart life capability needed to satisfy the space shuttle mission and reusability requirements. Use of additional ACS engines requires more impulse propellant because of a lower I_{sp} but eliminates the restart propellant losses. Further, no significant unstaballation or develop penalties are introduced.

A single systems for both attitude control and maneuvering is selected as the baseline because of its simplicity, minor development and installation effects and mission target without 2nd stage modifications.

PROPELLANT VOLUME UTILIZATION

107 FT HL-10



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GO-AROUND DEFINITION

Four approach and/or go-around options have been considered. They are landing assist, powered approach, 360° turn, and wave off. The effect of these options on payload is presented in the facing chart. These options may be utilized individually or in combination as indicated on the left side of the table. The resulting incremental changes in payload capability are summarized in the right hand column. The simplest option, landing assist, does not provide for go-around but merely intermittent glide slope control to minimize the probability of a go-around requirement. The second option, powered approach, provides energy to maintain an IFR glide slope. The 360° turn assumes sufficient cues to decide on a go-around requirement at 2000 feet altitude after which a 360° turn is performed to acquire a corrected approach pattern. The last option, wave off, considers a climb-out from 50 foot altitude after wave off with subsequent go-around and acquisition of the outer marker. This last option represents our baseline. Changes in payload capability resulting from alternate design options vary from (-) 3500 lbs to (+) 15,000 lbs.

GO-AROUND DEFINITION
Orbiter

OPERATIONAL OPTIONS				EFFECTS	
LANDING** ASSIST	POWERED** APPROACH	360° TURN AT 2000 FT	WAVE* OFF	THRUST/WT (REQ'D)	PAYLOAD INCREMENT (LB)
√				0.10	+ 16,800
	√			0.24	+ 7,500
	√	√		0.28	+ 4,000
	√		√	0.33	0 (BASELINE)
	√	√	√	0.33	− 3,500
√			√	0.33	+ 1,200

* CLIMB-OUT FROM 50 FT ALTITUDE AND RETURN TO OUTER MARKER

** WITH ENGINE OUT

ILRVS 498F

BOOST ENGINE OUT - ORBITER

In the event of a 2nd stage boost engine failure, the corresponding reduction in thrust to weight ratio requires an increase in the thrust deflection angle thus increasing the maneuvering ΔV losses. One method of compensating for this effect is to increase the vehicle ΔV capability at the expense of payload. Alternate methods, which do not effect payload, would be to call upon existing reserves (flight performance or orbit maneuver) or to provide increased thrust by designing the engine for an emergency overspeed capability.

Since the study ground rules were not specific as to how or why to provide an engine out capability, different combinations of these options were evaluated with regard to payload effects. Evaluation results are summarized on the opposite page.

For the baseline design, engine out capability is taken to mean the requirement to complete the mission without infringing on ΔV reserves already allocated for other contingencies. However, the engine is assumed to have a 25% emergency overspeed capability. This is considered the maximum allowable without incurring engine weight penalties. As indicated, a significant increase in payload could be obtained by assuming the availability of existing ΔV reserves in order to complete the mission. Similarly, the use of a safe abort to orbit criteria rather than mission completion would make available the entire orbit maneuver ΔV budget (less an allowance for de-orbit) and would also eliminate the need for any engine overspeed capability.

The significant payload effects associated with interpretation of the engine out requirement indicates the need for a more specific definition and must be considered when comparing the performance of different vehicle configurations.

BOOST ENGINE OUT
Orbiter

OPERATIONAL OPTIONS						EFFECTS	
NO RESERVES	FPR	FPR & OMR	SAFE ABORT	OVERSPEED		ΔV PENALTY* (FPS)	PAYLOAD (LB)
				0	25%		
V					V	590	25,000
	V				V	590	27,900
		V			V	590	32,500
			V	V		1,340	32,500
V				V		1,340	14,300

COMPLETE MISSION

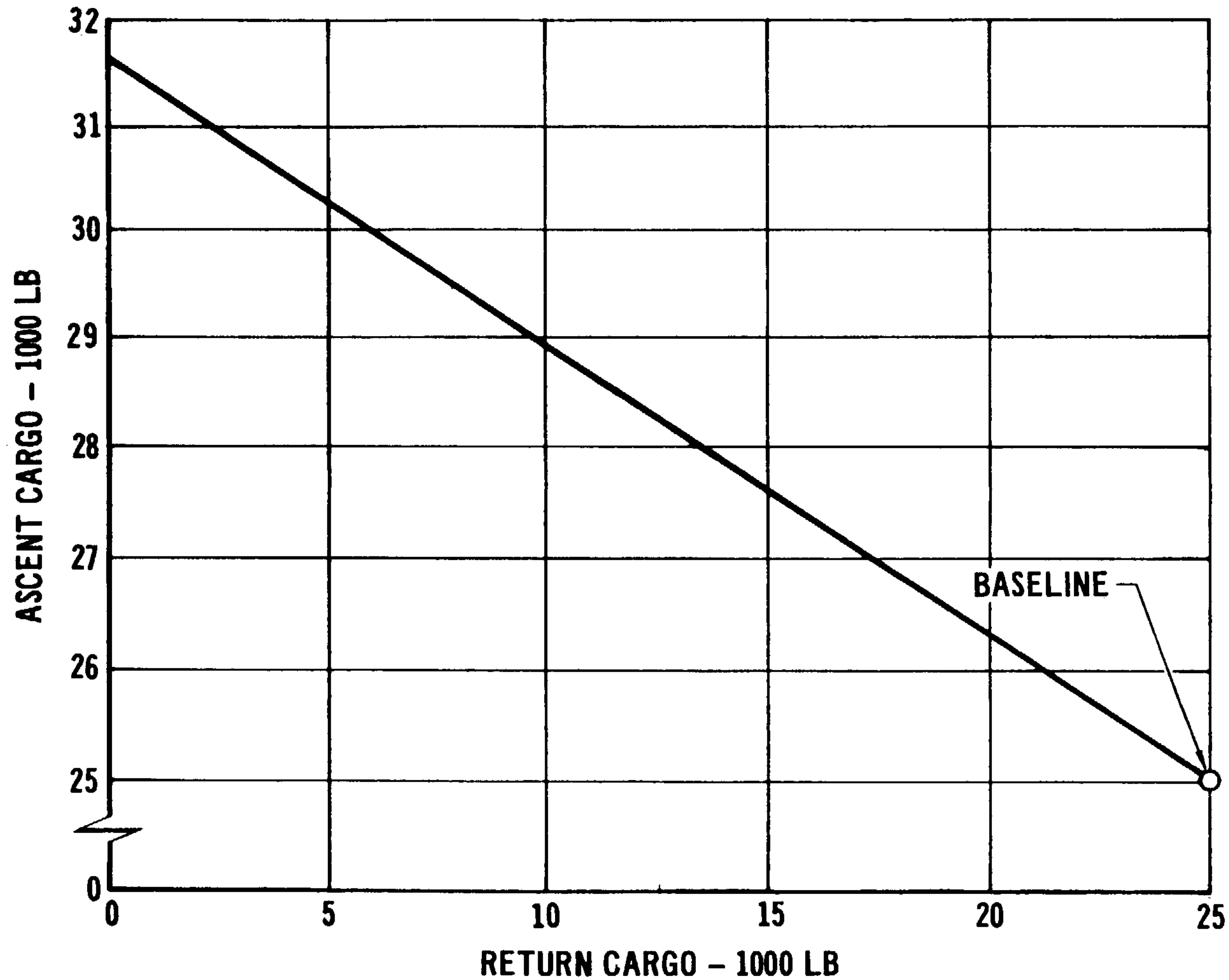
* WITH RESPECT TO NOMINAL PERFORMANCE

ILRVS 497F

EFFECT OF RETURN CARGO CAPABILITY

This chart illustrates the effect of return cargo on spacecraft weight. The baseline spacecraft was designed to return all the ascent cargo. If only a portion of this cargo is returned, the amount of ascent cargo could be increased as shown. This tradeoff study was conducted holding the total gross lift off weight constant. The increase in ascent cargo capability is a reflection of the weight reduction in those systems which are designed or influenced by retrograde, reentry and landing weight such as the retrograde and landing propulsion systems and the landing gear.

EFFECT OF RETURN CARGO CAPABILITY



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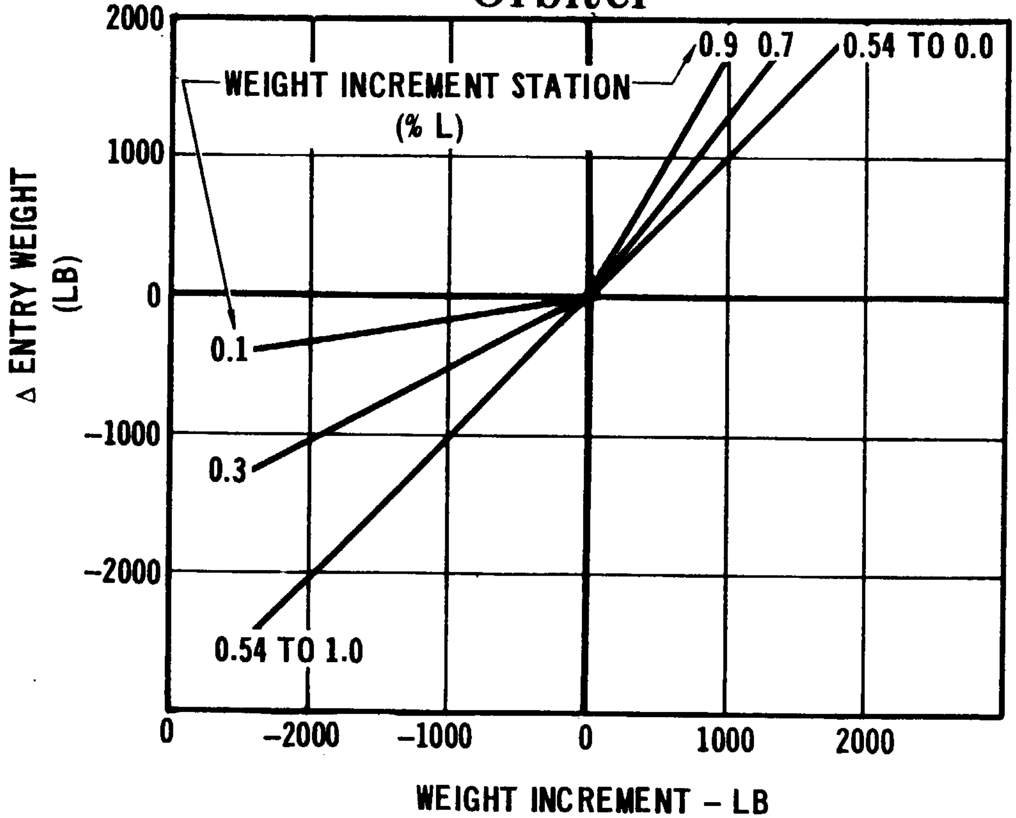
EFFECTS OF LOCALIZED WEIGHT INCREMENTS

This chart illustrates the importance of the location of a weight element which is to be added or removed from the vehicle. This condition results from the requirement to maintain the reentry c.g. at a specified body station. From the chart it is seen that any element whose c.g. is at the c.g. of the vehicle will be 100% effective. However, if an element is to be removed whose c.g. is in front of the desired location, it will be necessary to add back ballast to reestablish the desired condition. The weight saving will therefore be less than 100% effective. As an example, if 1200 lbs. is to be removed with a c.g. at .3L, it is necessary to add 630 lbs. of ballast resulting in only 47% effectiveness.

By the same token, if weight is added behind the desired c.g., it will be more than 100% effective, again because of the need to add ballast. The addition of 1200 lbs. at .7L results in a total weight increase of 1550 lbs. for an effectiveness of 129%.

EFFECTS OF LOCALIZED WEIGHT INCREMENTS
Orbiter



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DESIGN VELOCITY BUDGET

Mission selection establishes ideal velocity requirements. The total desgin velocity budget is composed of

- o Ideal mission velocity
- o Ascent phase losses
- o Contingency (Flight performance reserve)

Since losses constitute 15 to 20% of the total design budget, ascent trajectory optimization is mandatory. The ascent trajectory has been shaped to minimize losses within the constraints imposed by mission ground rules and configuration design.

The effect of relaxing the ascent acceleration constraint from 3g's to 4g's is to reduce losses by 40 ft/sec for the selected configuration. Total flight time at the constant 3g level is approximately 45 seconds during first stage operation and 55 seconds during second stage. For a 4g limit ground rule, 4g operation occurs only during second stage flight for less than 15 seconds.

VELOCITY BUDGET SENSITIVITY

Nominal Performance

	MAXIMUM AXIAL LOAD FACTOR 3G'S	4G'S
• INSERTION VELOCITY (45 X 100 N MI, i = 55°)	25,000	25,000
GRAVITY LOSSES	4,226	4,206
MANEUVER LOSSES	171	155
ALTITUDE THRUST LOSSES	340	340
DRAG LOSSES	694	690
• NOMINAL ASCENT PHASE BUDGET, ΔV_N	30,431	30,391
FLIGHT PERFORMANCE RESERVE (0.75 % ΔV_N	229	229
• TOTAL ASCENT PHASE BUDGET	30,660	30,620
ON-ORBIT BUDGET	2,000	2,000
• TOTAL ΔV BUDGET	32,660	32,620

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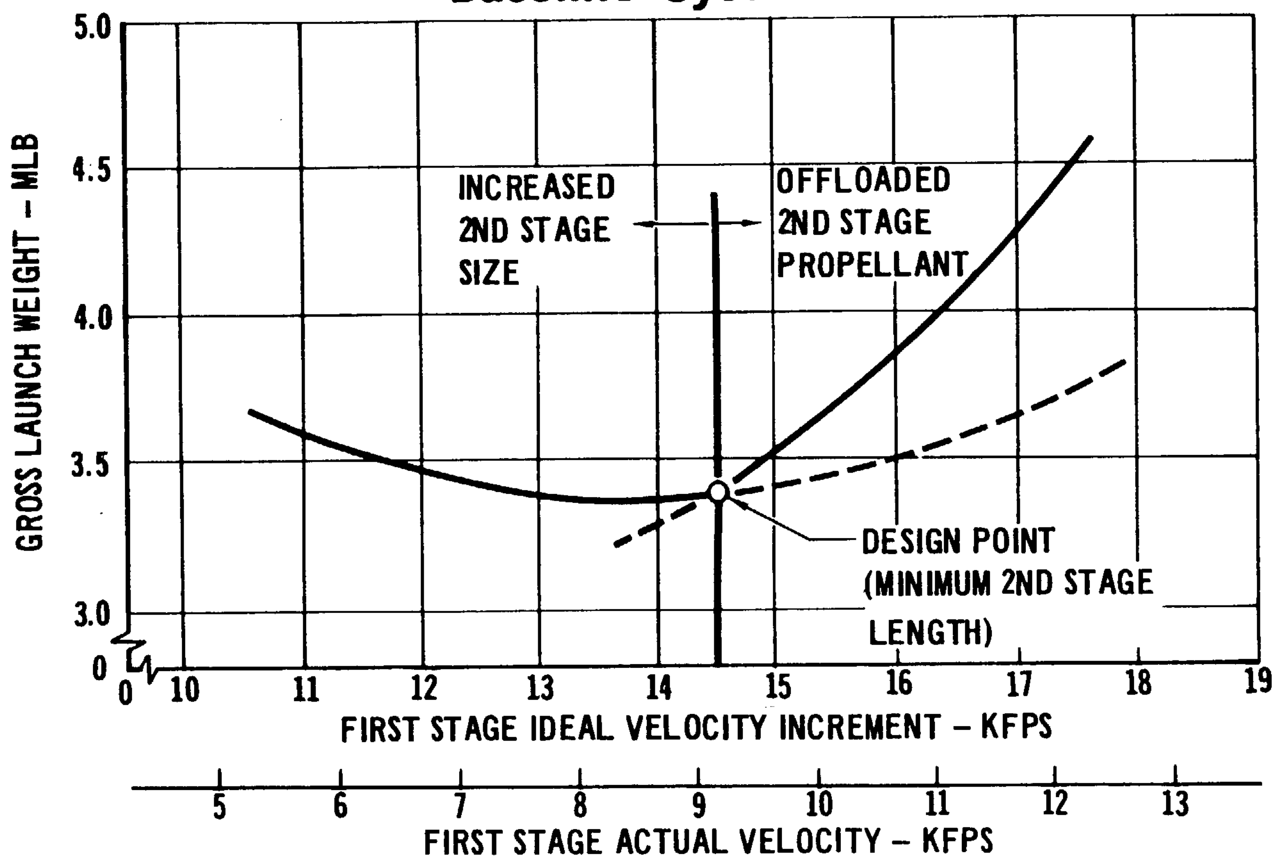
VELOCITY SPLIT DETERMINATION

The effect of velocity apportionment between stages on the vehicle gross launch weight is shown on the opposite page. The design point is predicated on use of the shortest length 2nd stage that will accommodate the 15 x 30 foot cargo canister. A maximum amount of boost propellant is loaded into the excess volume that exists even with a minimum length vehicle. The resulting 2nd stage ΔV is then subtracted from the total required to determine the 1st stage velocity increment. As indicated, this design point is very near the optimum for minimum gross launch weight. To increase the 1st stage ΔV requires off-loading the 2nd stage, an inefficient approach from the standpoint of volume utilization and hence gross launch weight. Decreasing the 1st stage ΔV necessitates an increased length second stage to obtain the additional ΔV required. Although a small weight reduction results, the increased cost of a larger 2nd stage plus the potential increase in the engine out ΔV penalty are expected to more than offset this weight advantage. Hence, the ΔV split for the baseline vehicle is that resulting from use of a minimum length 2nd stage.

VELOCITY SPLIT DETERMINATION
Baseline System



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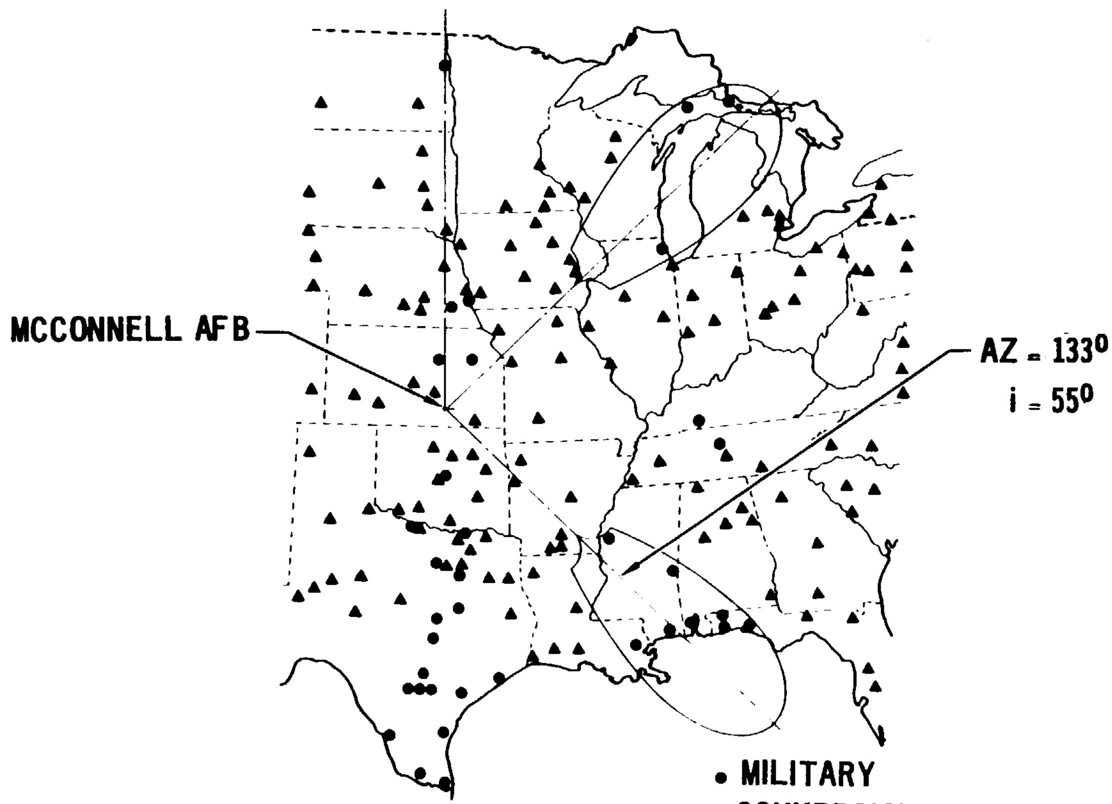
CARRIER LANDING SITES
(NO CRUISE REQUIRED)

The possibility of using an established Military or Commerical airfield located within the Continental United States as a launch site was investigated and the results shown on the opposite page.

McConnell AFB in Kansas was slected as a prime candidate for a launch site, although it is recognized that there are unresolved problems associated with populace safety and air traffic pattern interference. After launch from McConnell on any azimuth, the carrier vehicle can land at an established field with a zero cruise range capability. This is true for both normal and aborted flights. The figure shows the zero cruise range footprint for a 133 degree azimuth launch and indicates the number of major airfields covered by the footprint. This footprint can be rotated around the launch site to obtain the number of potential landing sites for any launch azimuth.

CARRIER LANDING SITES (NO CRUISE REQUIRED)



ILRVS-467F

SENSITIVITIES
(25,000 LB PAYLOAD)

Payload sensitivities to variation in these desing and performance parameters assume a constant gross launch weight and the gross launch weight sensitivities were based on constant payload (cargo) weight. These sensitivities, as partial derivatives of payload weight and gross launch weight with respect to the given parameters, vary with vehicle size and weight. This is borne out by comparison with the sensitivity chart for 50,000 lb payload.

SENSITIVITIES

REFERENCE:

P/L WEIGHT	25,000 LB
P/L SIZE	15' D x 30' L
GLOW	3.40 MLB

PARAMETER	CONST. GLOW $\frac{\Delta W_{PL}}{\Delta \text{ PARAMETER}}$	CONST. P/L $\frac{\Delta W_{GL}}{\Delta \text{ PARAMETER}}$
1ST STAGE INERT WT.	– 0.16 LB/LB	+ 6.5 LB/LB
2ND STAGE INERT WT.	– 1.0 LB/LB	+ 38.0 LB/LB
ΔV – LAUNCH	– 12.8 LB/FPS	+ 528 LB/FPS
ΔV – ORBIT	– 14.0 LB/FPS	+ 560 LB/FPS
PROPELLANT I_{SP}	+ 843 LB/SEC	– 35,400 LB/SEC
CRUISE RANGE	– 152 LB/NM	+ 5,790 LB/NM
GO AROUND (ORBITER)	– 377 LB/MIN	+ 12,400 LB/MIN
RETURN CARGO	– 0.27 LB/LB	+ 10.5 LB/LB

ILRVS — 385F

INCREMENTAL EFFECTS
(3.4 MLB)

The incremental effects on useful payload (cargo) resulting from specific modification in the design or operational modes of the basic two stage configuration, are shown in the chart. In assessing the net weight effect of each design or operational change, the basic vehicle dimensions were held constant. Also, the gross launch weight is held constant at 3.4 MLB in these calculations, and the net weight addition (+) or penalties (-) associated with the design or performance parameter changes are reflected solely in the orbiter payload (cargo). For example, if the carrier does not return to its launch site, 10,200 lbs. of cargo could be added as a result of the decreased propellant and other spacecraft systems. Similarly, if we were to incorporate a pumped idle mode during launch for the HL-10 rather than the nominal series burn, 5,700 lb. of cargo would be deleted. In assessing the net weight effect of each design or operational change, the basic vehicle dimensions were held constant. The .75% ΔV reserve which is nominally in the two stage configuration, amounts to about 230 fps. If this were deleted, an additional 3000 lb. of payload could be added.

INCREMENTAL EFFECTS

REFERENCE:

P/L WEIGHT	25.000 LB
P/L SIZE	15'Dx 30'L
GLOW	3.40 MLB

Δ EFFECT	CONST. GLOW ΔW_{PL}	CONST. P/L ΔW_{GL}
• NO FLYBACK TO LAUNCH SITE	+10,200	−312,000
• SERIES BURN TO IDLE MODE	− 5,700	+283,000
• PARALLEL BURN WITH CROSS FEED	+ 3,200	−124,000
• NO 1ST STAGE INERT CONTINGENCY	+ 6,400	−235,000
• NO 2ND STAGE INERT CONTINGENCY	+14,000	−524,000
• NO INERTS (BOTH STAGES) CONTINGENCY	+20,500	−695,500
• NO 0.75% ΔV RESERVE	+ 3,000	−124,000
• H_2 CRUISE FUEL	+ 6,210	−252,000

NOTE: INCREMENTS ARE NOT ADDITIVE

ILRVS-473F

EFFECT OF PAYLOAD WEIGHT AND SIZE

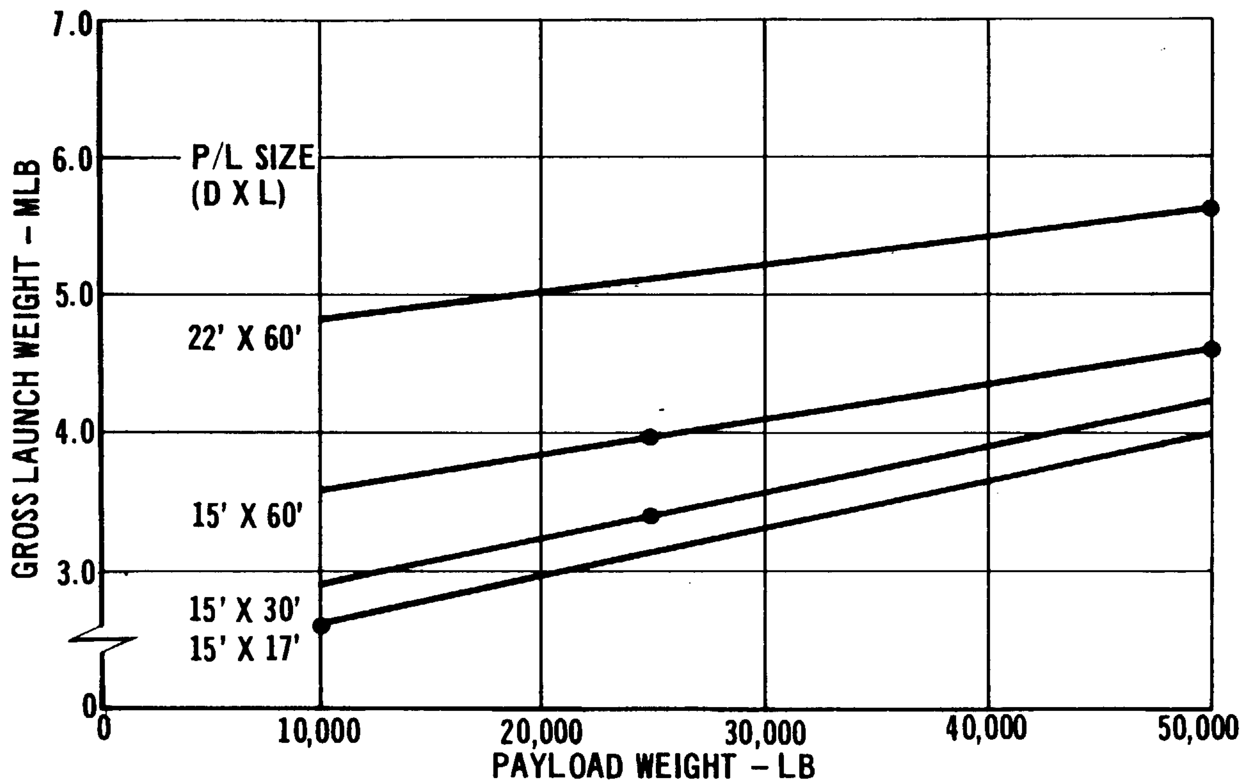
The gross launch weight variation as a function of payload weight change is presented on the accompanying chart for four vehicles. Each of the vehicles is point designed for a specific payload size and weight capability. The payload size (diameter and length) is given in feet at the left of each curve. The 15 x 17 vehicle is designed for 10,000 lbs. payload while the 15 x 30 and 15 x 60 use 25,000 lbs. and the 22 x 60 is 50,000 lbs.

The curve illustrates that a larger vehicle suffers a smaller weight change in gross weight per pound of payload than one designed for a smaller payload. The 15 x 17 vehicle increases at the rate of about 44 lbs/lb payload but the 22 x 60 only increases at the rate of about 20 lb/lb payload.

Note, however, that the payload geometry has a greater effect than payload weight. This is shown by the 15' x 60' payload which has a larger gross launch weight than the same payload in a 15' x 30' container.

The five circled points on this chart represent the four alternate payloads which were analyzed as required by the study contract.

EFFECT OF PAYLOAD WEIGHT AND SIZE



ILRVS–427F

SENSITIVITIES
(50,000 LB PAYLOAD)

Payload sensitivities to variations in these design and performance parameters assume a constant gross launch weight while the gross launch weight sensitivities were based on constant payload weight. These sensitivities vary with vehicle size and weight.

SENSITIVITIES

REFERENCE:

P/L WEIGHT	50,000 LB
P/L SIZE	15' x 60' L
GLOW	4.55 MLB

PARAMETER	CONST. GLOW $\frac{\Delta W_{PL}}{\Delta \text{ PARAMETER}}$	CONST. P/L $\frac{\Delta W_{GL}}{\Delta \text{ PARAMETER}}$
1ST STAGE INERT WT.	– 0.16 LB/LB	+ 5.5 LB/LB
2ND STAGE INERT WT.	– 1.0 LB/LB	+ 32.6 LB/LB
ΔV – LAUNCH	– 18.7 LB/FPS	+ 614 LB/FPS
ΔV ORBIT	– 21.0 LB/FPS	+ 690 LB/FPS
PROPELLANT I_{SP}	+ 1,262 LB/SEC	– 41,200 LB/SEC
CRUISE RANGE	– 180 LB/NM	+ 5,860 LB/NM
GO AROUND (ORBITER)	– 499 LB/MIN	+ 16,200 LB/MIN
RETURN CARGO	– 0.27 LB/LB	+ 9.0 LB/LB

ILRVS-384F

INCREMENTAL EFFECTS
(4.55 MLB)

The effects on payload (cargo) weight of various design and operational changes to the basic 4.55 MLB two stage configuration, are shown on this chart. The gross launch weight and vehicle dimensions are held constant in the derivation of these payload weight changes, and a + or - accompanying the ΔW_{PL} indicates payload increase or decrease respectively. The values shown differ slightly from those presented for the 3.4 MLB configuration, and are generally of a greater magnitude. The only exceptions to this are the flyback and series burn increments. The staging velocity of the 4.55 MLB vehicle is less than the 3.4 which results in a shorter distance to flyback and therefore less sensitivies. In the case of series burn each second stage expends approximately the same propellant. However, this is a smaller percentage of the total in the 4.55 MLB vehicle and again results in less senstivity.

INCREMENTAL EFFECTS

REFERENCE:
P/L WEIGHT 50,000 LB
P/L SIZE 15' D x 60' L
GLOW 4.55 MLB

Δ EFFECT	CONST. GLOW ΔW_{PL}	CONST. P/L ΔW_{GL}
• NO FLYBACK TO LAUNCH SITE	+ 9,800	−268,000
• SERIES BURN TO IDLE MODE	− 5,700	+211,000
• PARALLEL BURN WITH CROSS FEED	+ 4,700	−144,000
• NO 1ST STAGE INERT CONTINGENCY	+ 7,700	−229,000
• NO 2ND STAGE INERT CONTINGENCY	+20,000	−615,000
• NO INERTS (BOTH STAGES) CONTINGENCY	+28,000	−787,000
• NO 0.75% ΔV RESERVE	+ 4,400	−144,000
• H_2 CRUISE FUEL	+ 6,000	−204,000

ILRVS-474F

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CONCEPTUAL DESIGN DEFINITION

Task 2

ILRVS-430F

GENERAL ARRANGEMENT HL-10

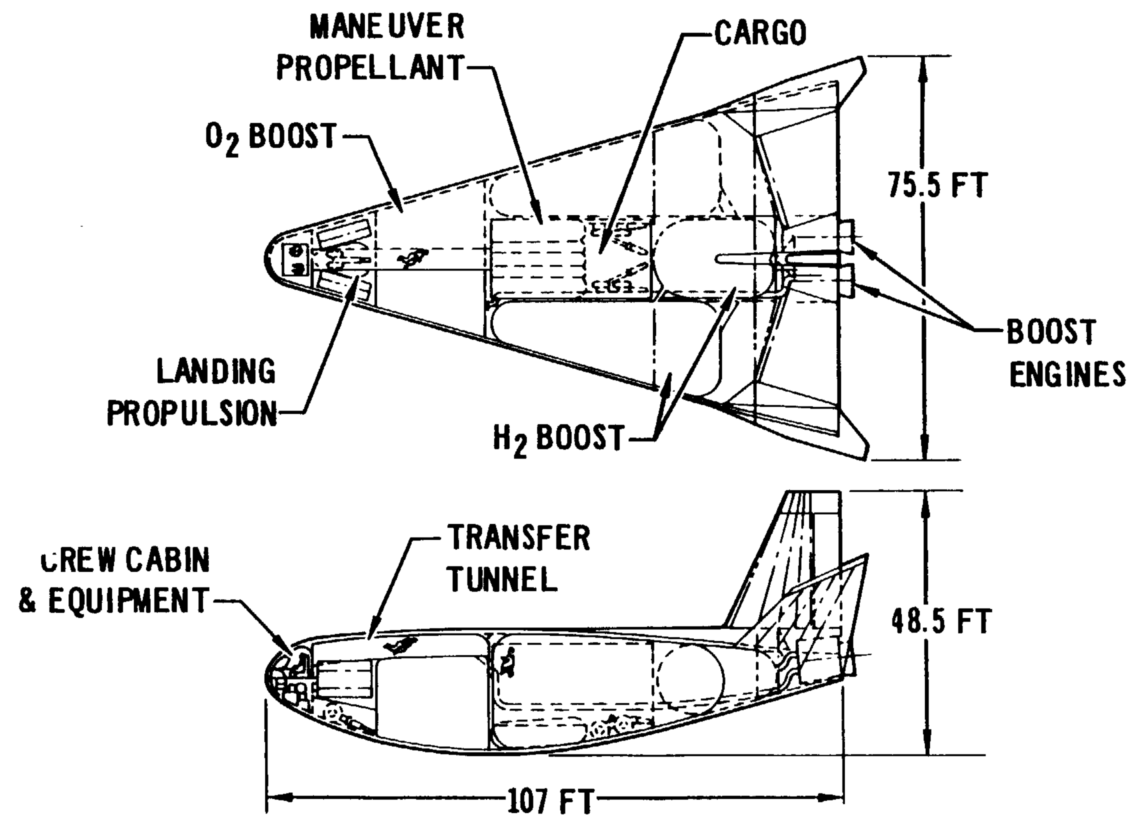
The structural concept and internal arrangement of the HL-10 second stage are shown here.

The cargo bay shows the 15 ft. dia. 30 ft. long container with 1 ft. allowed at either end and on the diameter for installation clearance and mounting provisions. The boost propellant oxidizer tank is forward of the cargo and a hydrogen tank is on either side of, and aft of, the cargo. The walls of these tanks are made to conform to the inner moldline of the vehicle whenever possible. The tank walls then become the primary load carrying skin for vehicle loads. The inner moldline skin forms an extension of the tank walls forward of the oxygen tank, between oxygen and hydrogen tanks and aft of the hydrogen tanks.

The forward compartment encloses crew cabin, avionics, power supply, nose gear and landing propulsion system. The landing engines are deployed out the sides of the vehicle for operation. Additionally, a tunnel is provided between the crew cabin and cargo bay to permit transfer of the crew to a cargo container during orbit operations. This tunnel is inside the moldline and on the vehicle center line above the oxygen tank.

Propellant for 2000 fps in-orbit maneuvering capability is provided by two tanks below the forward portion of the cargo bay. The main landing gear is positioned on either side below the aft portion of the cargo bay. Thrust loads from the 2 boost engines are transferred to propellant tank walls/body skin by a lateral beam.

GENERAL ARRANGEMENT HL – 10



ILRVS-262F

HL-10 INTERIM ARRANGEMENT

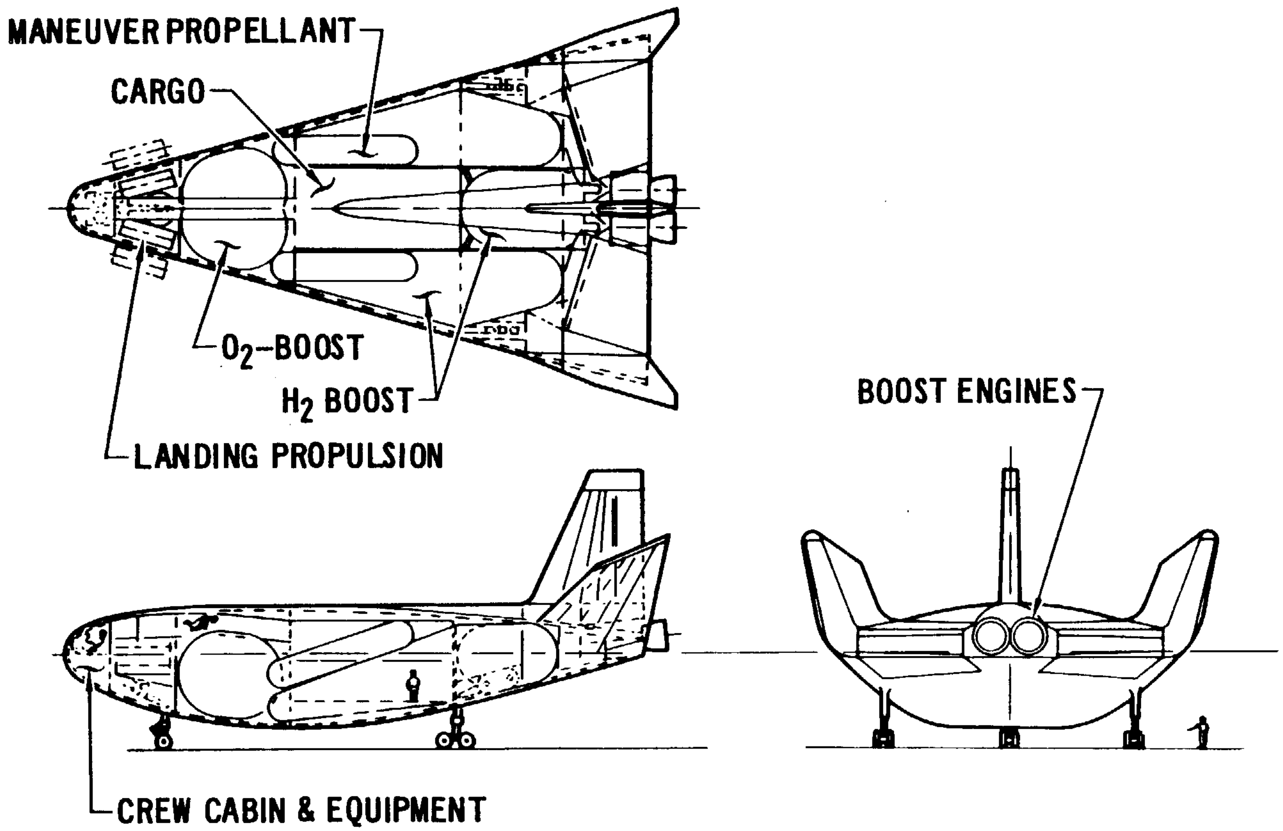
The arrangement shown was used as an orbiter baseline for subsystems, performance and stress analysis during the early phase of the two stage recoverable spacecraft study.

This arrangement utilized structural full depth webs on either side of the cargo bay to transfer the launch thrust loads into the vehicle skin. An oxidizer tank forward of the cargo and hydrogen tanks on either side of and aft of the cargo provide the boost propellants. Maneuvering propellant is contained in tanks on either side of the cargo, above the main hydrogen tanks.

The spacecraft volume forward of the main oxygen tank includes provisions for crew cabin, power supply, avionics and landing systems. Turbojet engines for landing assist are deployed out either side of the spacecraft. The propellant for these engines is located in a tank between the stowed engines. A transfer tunnel inside the spacecraft connects the crew cabin and cargo container.

Investigation of this spacecraft concept was terminated at the initiation of the integral tank concept study. The results of subsystem studies and spacecraft volume utilization, however, were carried from this concept to the latter one.

HL-10 INTERIM ARRANGEMENT

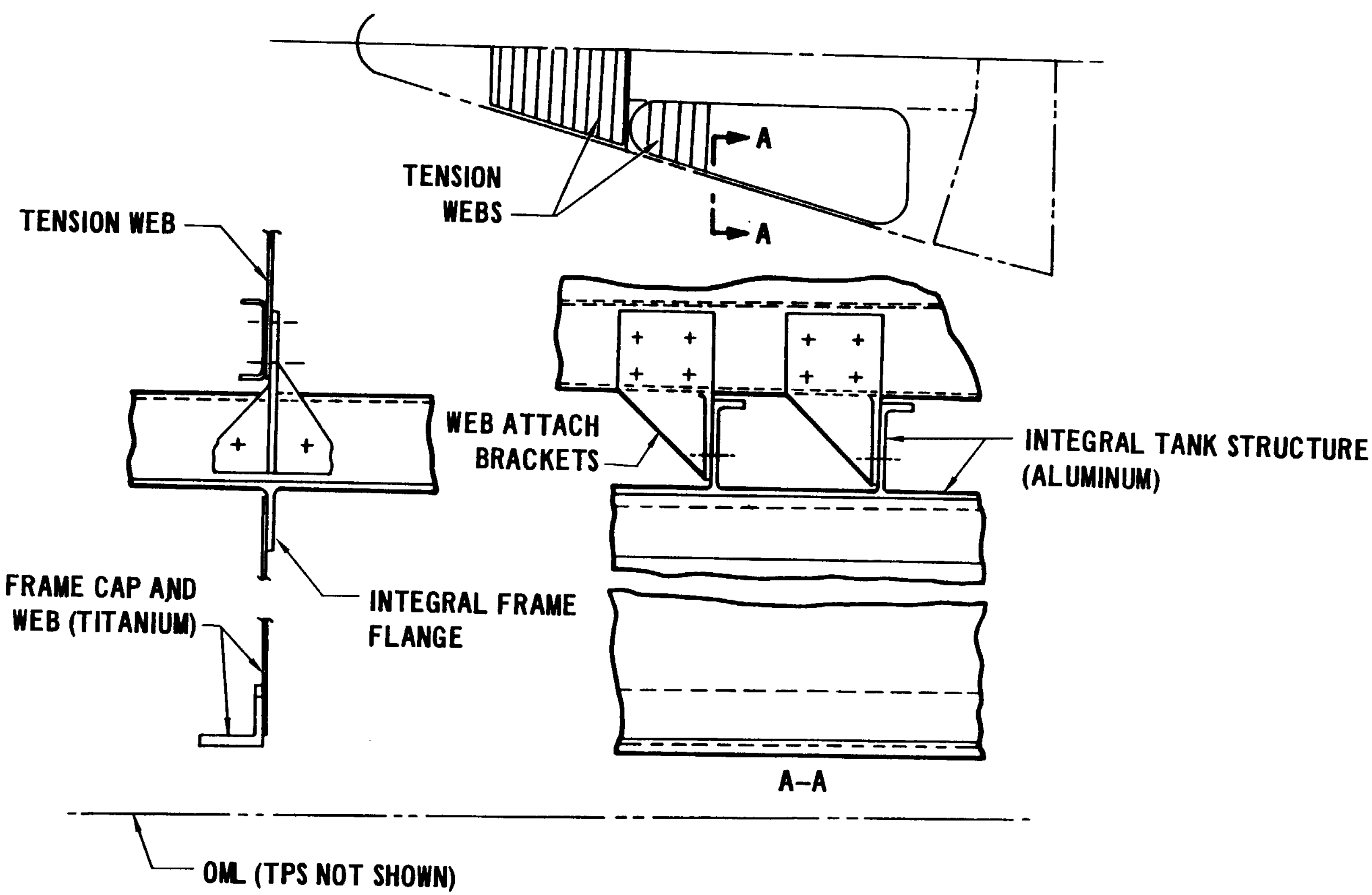


ILRVS–263F

HL-10 ORBITER INTEGRAL TANK STRUCTURAL CONFIGURATION

The integral side wall structural arrangement contains internal stringers at 2.0 inch spacing with ring flanges at 15 inch spacing. The integral skin/stringers provide a capability for carrying body loads and in addition distribute internal pressure loads to bi-axially loaded internal baffle/webs. External frames support the surface thermal protection and distribute dynamic pressure loads to the shell structure. The frame web and outboard cap are made of titanium to minimize conductance of heat to the inner structure. Space between the tank structure and outer mold line contains fibrous insulation with a minimum two inch void maintained for purging this space prior to launch.

STRUCTURAL CONFIGURATION
ORBITER INTEGRAL TANK



ILRVS–252F

INTEGRAL CRYOGENIC TANK/STRUCTURE CONCEPTS

A comparison of structural and cryogenic insulation arrangements was made for the integral tanks to determine the relative merits of the various concepts. This study considered both internal and external insulation of the cryogenic hydrogen tank, plus primary structure external to, internal to, and on both sides of the cryo tank wall.

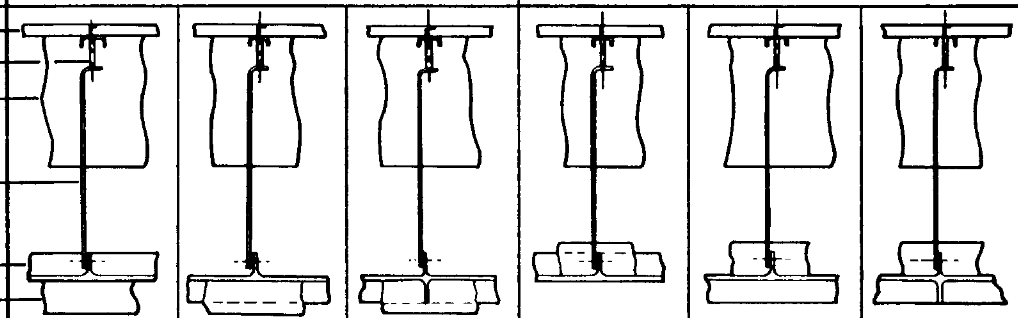
The structural arrangement with both rings and stringers outside the tank wall provide the maximum capability for support of external structure. The internal rings and stringers, however, provide the maximum ability for attaching internal structure and equipment.

Internal cryogenic insulation generally provides the minimum heat leak paths into the cryogenic fluid. The external cryo insulation, which might be a fibrous type, requires a tank skin temperature of -423°F and mechanical support. Internal insulation permits a higher tank wall temperature. External insulation also requires a vapor barrier to prevent frost and moisture build-up between the structure and insulation.

The structural fabrication becomes most complex for the concepts where all the primary support structure is on one side of the tank wall because secondary structure must also be provided on the opposite side. Fabrication of the cryogenic insulation installation is more difficult for the mechanically supported external insulation. Inspection and repair of insulation and structure is most complex with both items external because the thermal protection panels and insulation must be removed to provide accessibility. The internal insulation may be maintained from within the tank by opening an access hatch. Structure external to the tank must still be inspected by removing moldline panels but does not require disruption of the cryogenic insulation.

An overall summation of relative merits of the various concepts indicates the first two cases as being most effective in terms of reducing the various penalties discussed. The choice between all external structure and divided structure, with internal insulation, then becomes a function of the amount of internal hardware required.

INTEGRAL CRYOGENIC TANK/STRUCTURE CONCEPTS

CRYO INSULATION	INTERNAL			EXTERNAL		
RINGS/STRINGERS	EXT	EXT/INT	INT	EXT	EXT/INT	INT
M. PANEL SUPPORT INSULATION SUPPORT RING TANK STRUCTURE CRYO INSULATION						
PROVISION FOR EXT STRUCTURE & TPS (1 = LEAST PENALTY)	1	2	3	1	2	3
PROVISION FOR INT WEBS & BAFFLES (1 = LEAST PENALTY)	3	2	1	3	2	1
STRUCTURE HEAT LEAKS (1 = LEAST PENALTY)	1	2	3	6	5	4
MECHANICAL CRYO INSUL SUPPORT (0 = NOT REQ, 1 = REQ)	0	0	0	1	1	1
COLD STRUCTURE (0 = NO, 1 = YES)	0	0	0	1	1	1
MOISTURE BETWEEN CRYO INSULATION & STRUCTURE (1 = MIN PROBLEM)	1	1	1	2	2	2
STRUCTURE BUILD-UP COMPLEXITY (1 = LEAST PENALTY)	2	1	3	2	1	3
INSULATION INSTALLATION COMPLEXITY (CRYO) (1 = LEAST PENALTY)	1	2	3	6	5	4
INSPECT AND REPAIR DIFFICULTY (1 = LEAST PROBLEM)	1	2	3	6	5	4

ILRVS-260F

HL-10 PROPULSION SYSTEMS

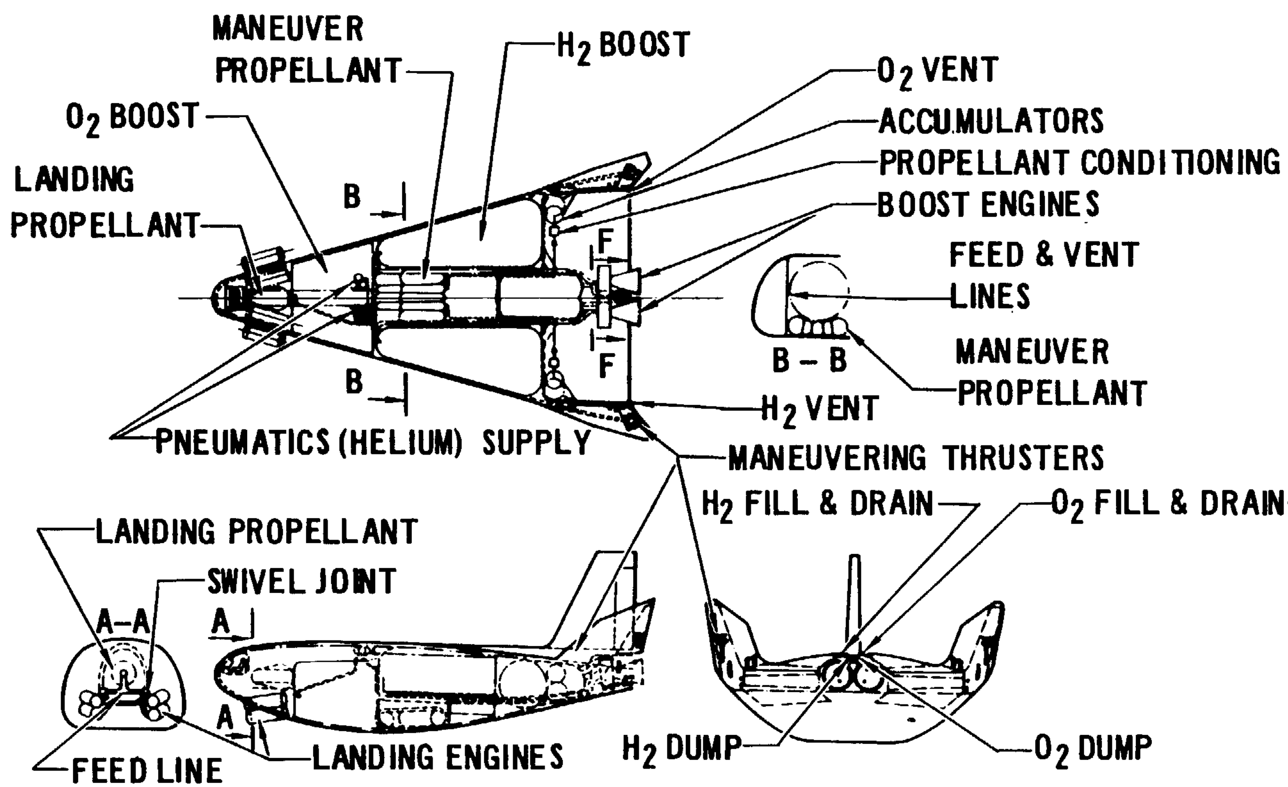
The propulsion systems shown provide boost, orbit maneuvering, and landing assist capability.

The boost propellant system includes a 6500 cu. ft. oxygen tank and 17500 cu. ft. of hydrogen in three tanks. The hydrogen tanks are inter-connected with large crossover lines to make the system operate as one tank for propellant feed. This propellant combination provides a 6 to 1 mixture ratio for the 2 448000 lb sea level thrust engines. Separate feed ducts are routed from the storage tanks to each engine. Tank fill is accomplished through the feed ducts with disconnects at the base of the vehicle.

The attitude control system and orbit maneuvering system use hydrogen and oxygen propellants which are stored cryogenically in a dual propellant tank located below the cargo bay. Separate but interconnected propellant conditioning systems are positioned on either side of the vehicle. 4000 lbs. thrust attitude control thrusters are mounted in the two tip fins. Roll control is provided by coupled thrusters. Pitch and yaw are provided by thrusters firing in one direction to provide the proper moments. Translation is fore-and-aft directions is supplied for incremental station keeping requirements in orbit.

Powered landing capability is supplied by 4 turbo-jet engines. These are deployed out the sides of the vehicle. They are deployed in a canted position with the forward end down to align their centerlines with airflow at an angle of attack. JP-4 fuel is contained in a tank between the stowed engines.

HL-10 PROPULSION SYSTEM



ILRVS–270F

HL-10 EQUIPMENT

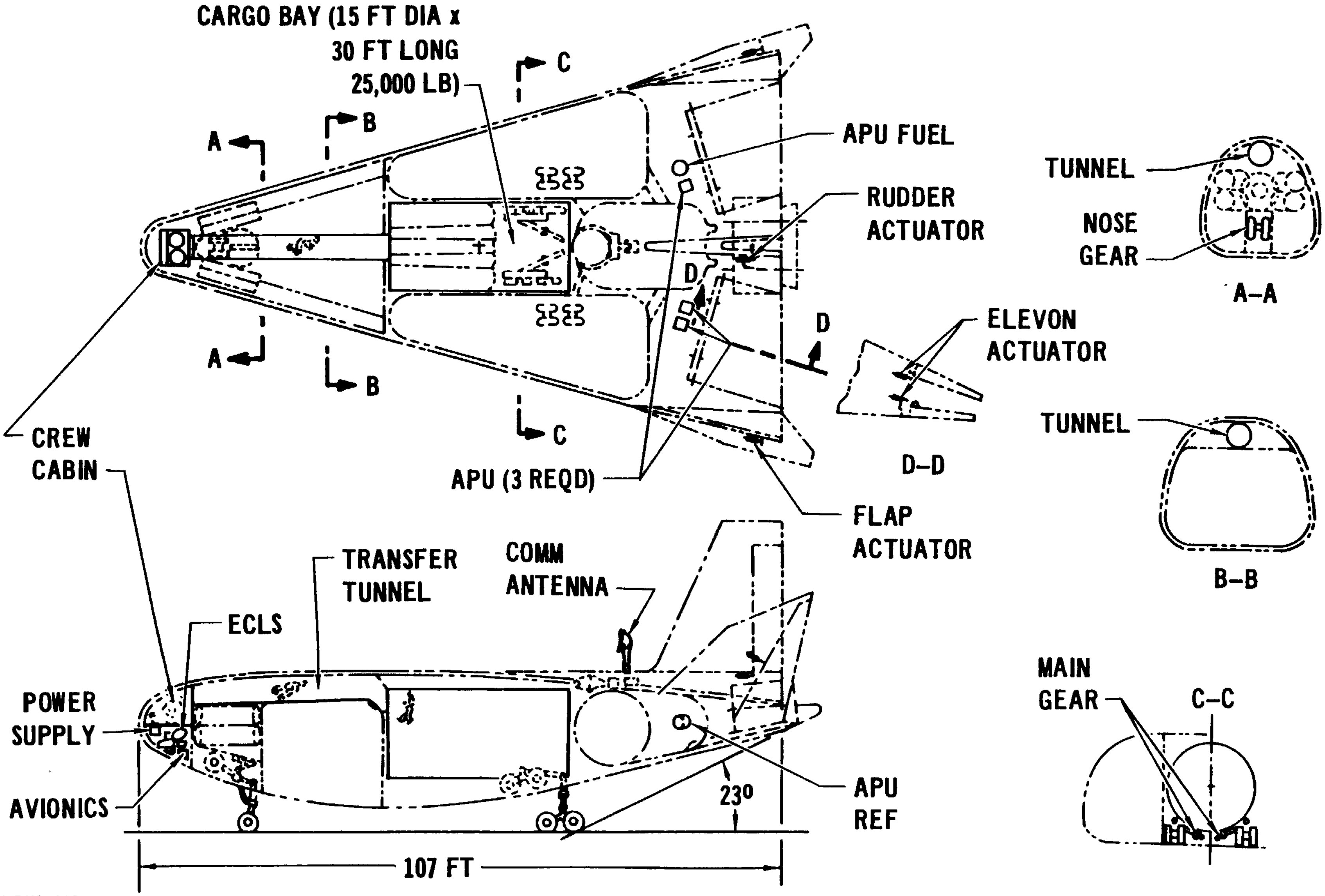
The equipment shown includes the spacecraft systems not defined by structure and propulsion subsystems.

The pressurized crew cabin is sized for a two man crew. They function in a shirtsleeve environment provided by a two gas (O_2-N_2) environmental control and life support system which occupies 50 cu. ft. A transfer tunnel from crew cabin to the cargo container is also provided with a controlled atmosphere. The cargo container is deployed by opening doors on the vehicle upper surface and mechanically translating the cargo up for accessibility to a space "tug".

Avionics and power supply are located as far forward in the vehicle as possible for c.g. considerations because of their relatively high density. Spacecraft avionics equipment occupies 40 cu. ft., part of which is included in the crew cabin as controls and displays. Also a 6 ft. dia. SHF dish antenna is located near the upper surface aft of the cargo bay. Power supply (batteries and fuel cells) for the electrical load requires 33 cu. ft. Power for the aero control system is provided by three auxillary power units using hydrazine fuel. This fuel is contained in an 8 cu. ft. sphere and the system is located in the aft end of the vehicle near the point of usage. Dual redundant hydraulic actuators impart the required rotational control to 2 tip fin flaps, 4 elevons and the rudder.

The main landing gear is located just aft of vehicle c.g. with the tire size and number determined by landing loads and concrete runway requirements. Deployment of the gear, with an 18 in. load deflection stroke, provides clearance for a maximum 23° angle of attack at touchdown.

HL-10 EQUIPMENT

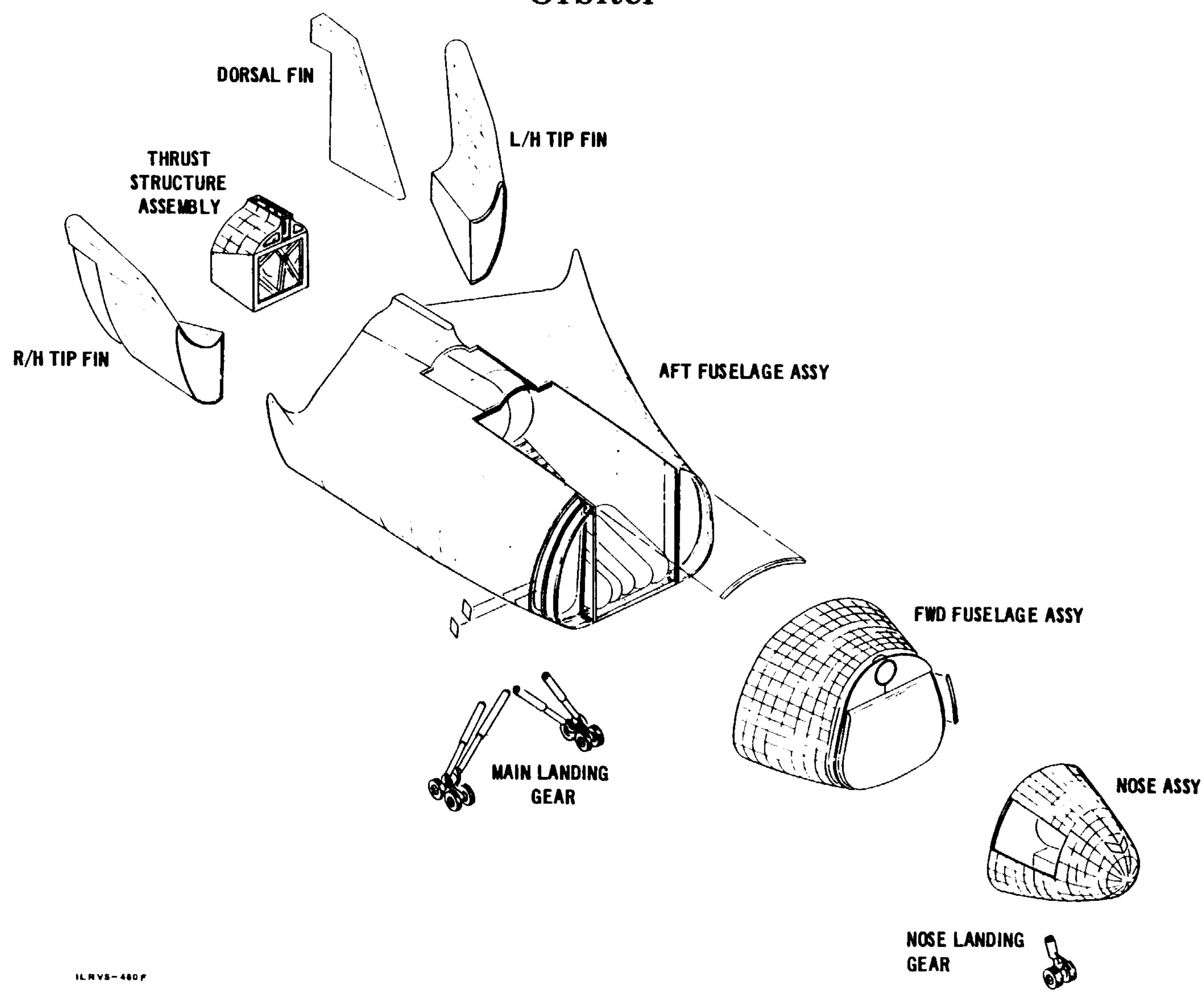


ILRVS-265F

PICTORIAL FLOW CHART - ORBITER

Major structural components shown for the HL-10 are the crew compartment, forward tank section, cargo and hydrogen tank section, thrust structure, and fins. The crew compartment contains the nose gear, jet engines and JP-4 fuel tank, and a section of the crew access tunnel. The forward tank section has the remaining portion of the access tunnel passing over the LO_2 tank. This tunnel leads from the crew compartment to the cargo container. The main fuselage section contains two LH_2 tanks, one on each side, two aft LH_2 tanks, the cargo compartment and maneuver propellant tanks. Fins and the thrust structure, which houses two boost engines, make up the remaining structure.

PICTORIAL FLOW CHART
Orbiter

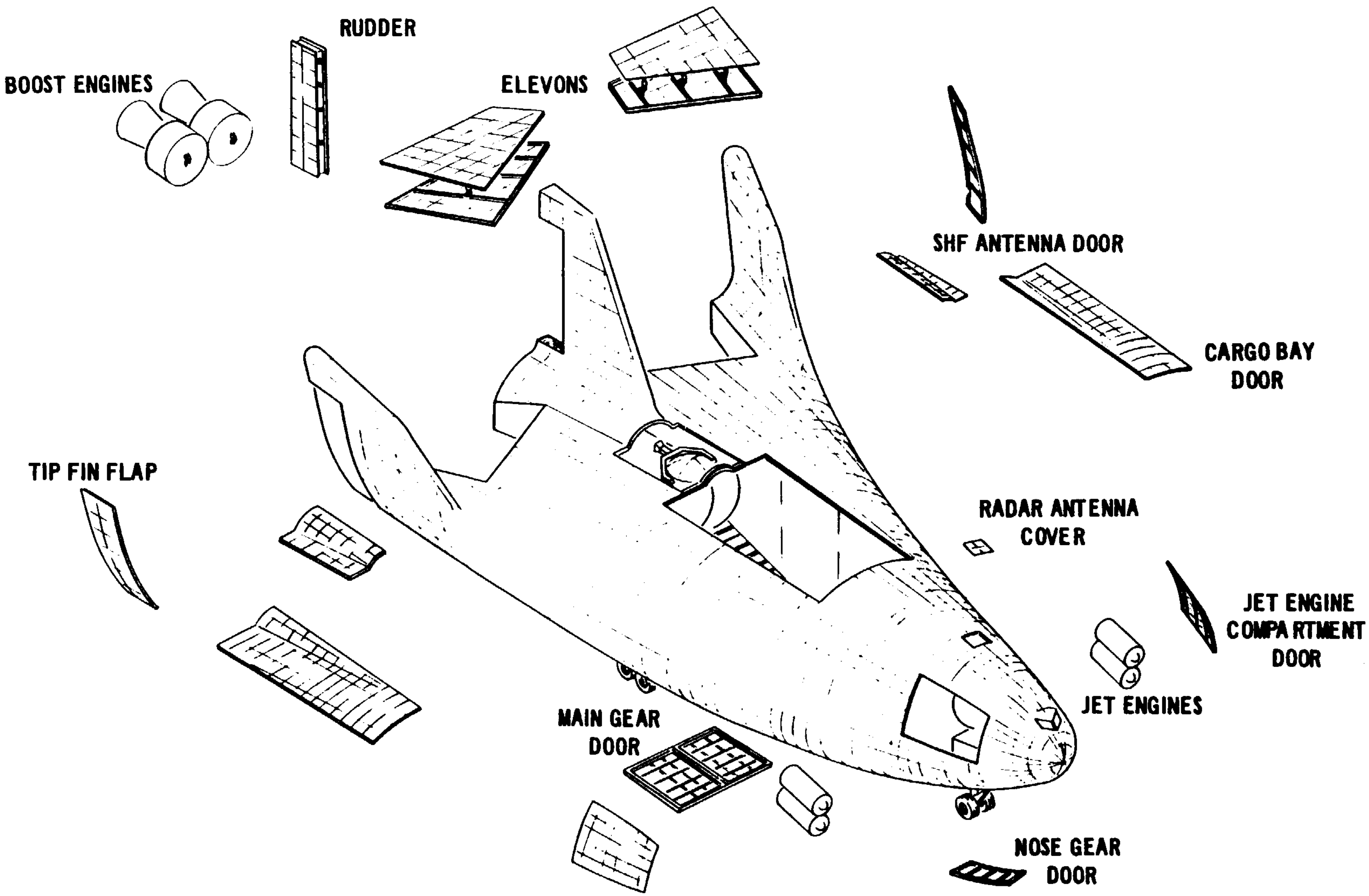


ILRVS-460F

PICTORIAL FLOW CHART - ORBITER

The HL-10, an integral tank/fuselage shell structural concept is shown partially assembled with jet engine and cargo compartment doors removed. The opening immediately aft of the cargo compartment houses the six-foot diameter dish antenna which retracts to a position over the aft LH_2 tanks. This vehicle, like the carrier, has light-weight aluminum primary structure which is sufficiently inboard and sufficiently insulated so that it reaches a maximum temperature of approximately 200°F.

PICTORIAL FLOW CHART
Orbiter



ILRVS 479F

HL-10 FERRY CONFIGURATION

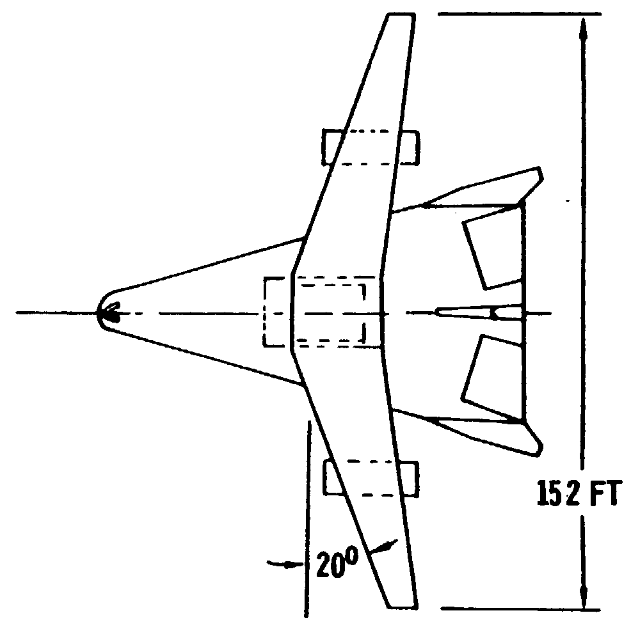
The HL-10 ferry configuration provides the operational capability of flying the vehicle from a remote site to the launch site or other location.

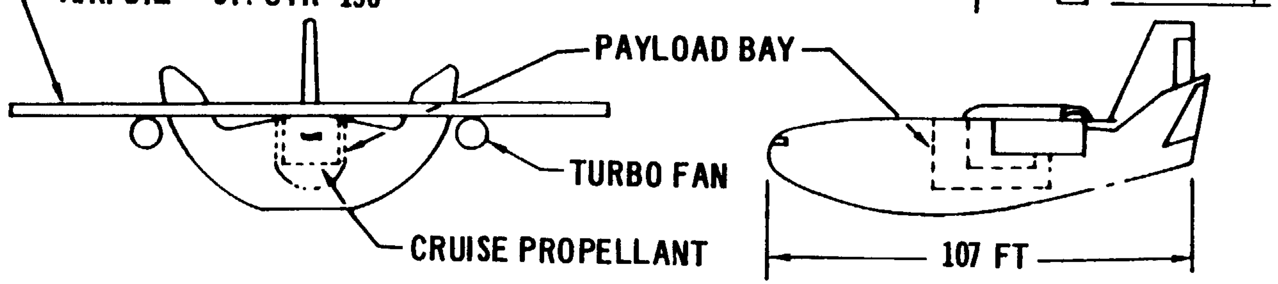
The HL-10 orbiter configuration has its cargo bay doors removed and an assembled package of wing, cruise engines and cruise propellant is attached with the propellant tank in the cargo bay. The wing has a ST.CYR 156 airfoil with a 20° sweep angle, 152 ft. span and .3 taper ratio. The basic HL-10 control system is used for aerodynamic control. This configuration was chosen to improve the HL-10 cruise capability ($L/D_{subsonic} = 4.1$). Similar configurations have been wind tunnel tested at LRC and shown to be feasible.

Two turbofan engines are mounted on the lower surface of the wing to provide the required cruise thrust. These engines are the same as the cruise engines on the first stage. Turbofan engines were selected to take advantage of their superior specific fuel consumption. The system for engine control is the only physical interface required between the HL-10 and ferry kit, aside from the structural interface. The existing vehicle structure will accept the kit installation. No landing gear penalty is imposed as the take-off weight of this configuration is about 20,000 lb. greater than normal landing weight from orbit.

HL-10 FERRY CONFIGURATION

MAX L/D (TRIM) = 8.5 (LRC TEST DATA)
FERRY KIT DRY WT = 31,000 LB
FERRY PROPELLANT WT = 19,000 LB
ENGINE THRUST (TF39) = 45,000 LB (SLS)
FERRY RANGE = 330 N MI





ILRVS-311F

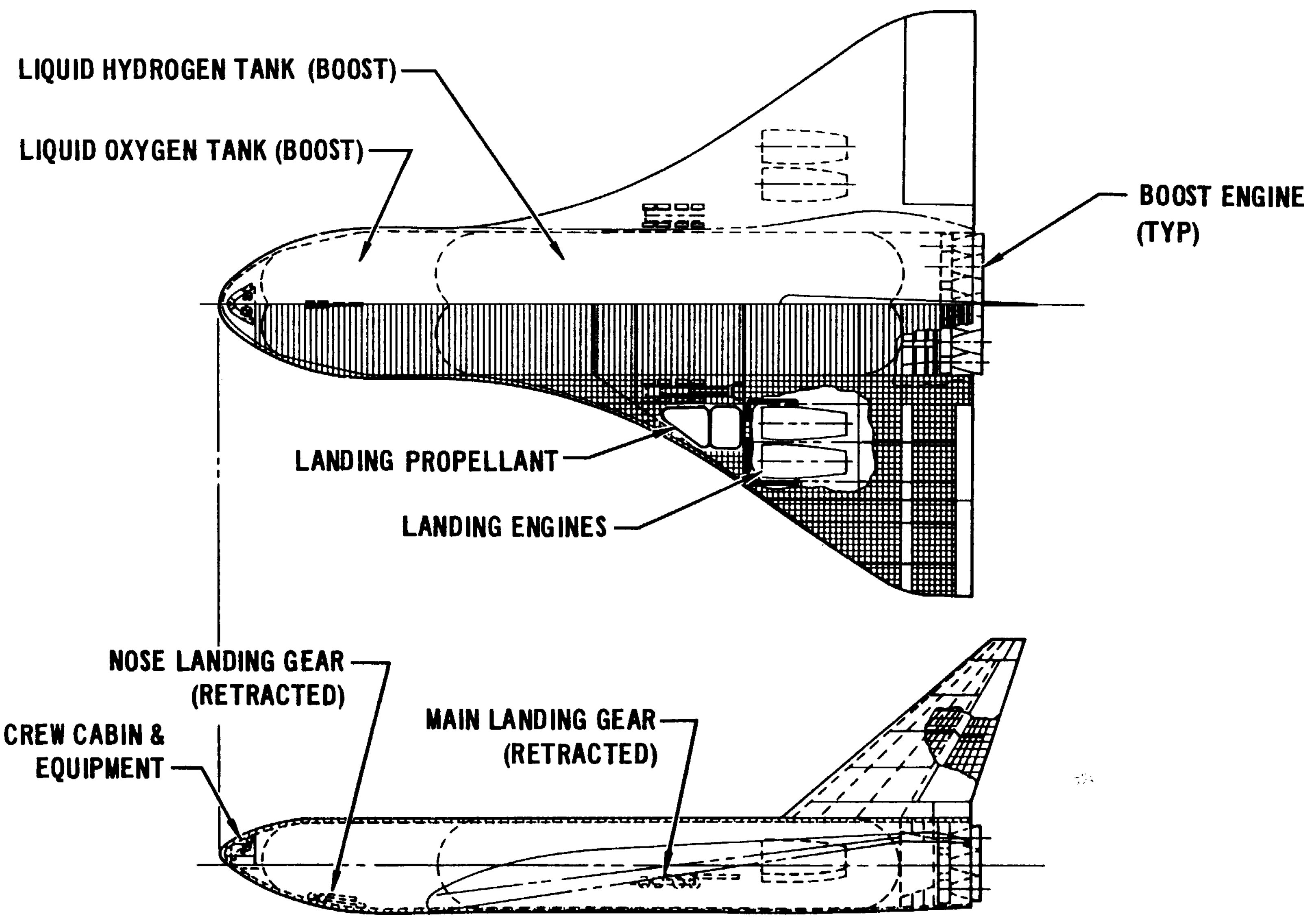
GENERAL ARRANGEMENT CARRIER

The structure and subsystems arrangement for the carrier are shown here. The planform view shows the external configuration on the right hand side of center-line and the internal arrangement on the left side.

The vehicle body contains a dual lobe propellant tank with the oxidizer forward and the hydrogen in the aft portion. The walls of this tank form the primary structural skin for the vehicle body. The forward end of the body is formed by an extension of these tank walls and provide a transition to the nose radius. This volume encloses the crew cabin, avionics, power supply, and the nose gear. The aft end of the body, housing the boost engines, thrust structure, and propellant utilization system is also an extension of the propellant tanks. The lower surface boattail at the aft end, and the raised nose, provides a negative camber body. A center line web between the two tank lobes, body rings and stringers complete the body structure.

The wing conforms to a modified Clark Y airfoil. A thick wing and low wing position was selected. This permits enclosing the landing engines, landing propellant system, and main gear in the wing and eliminates the need for separate fairings on the body or wing to enclose these systems. The forward wing spar lies along a constant per cent of the chord. The other spars are normal to the body sides. This provides a transition to the body rings and a load path for wing carry-through without the necessity for penetrating the propellant tank walls with primary structure.

GENERAL ARRANGEMENT CARRIER



ILRVS-264F

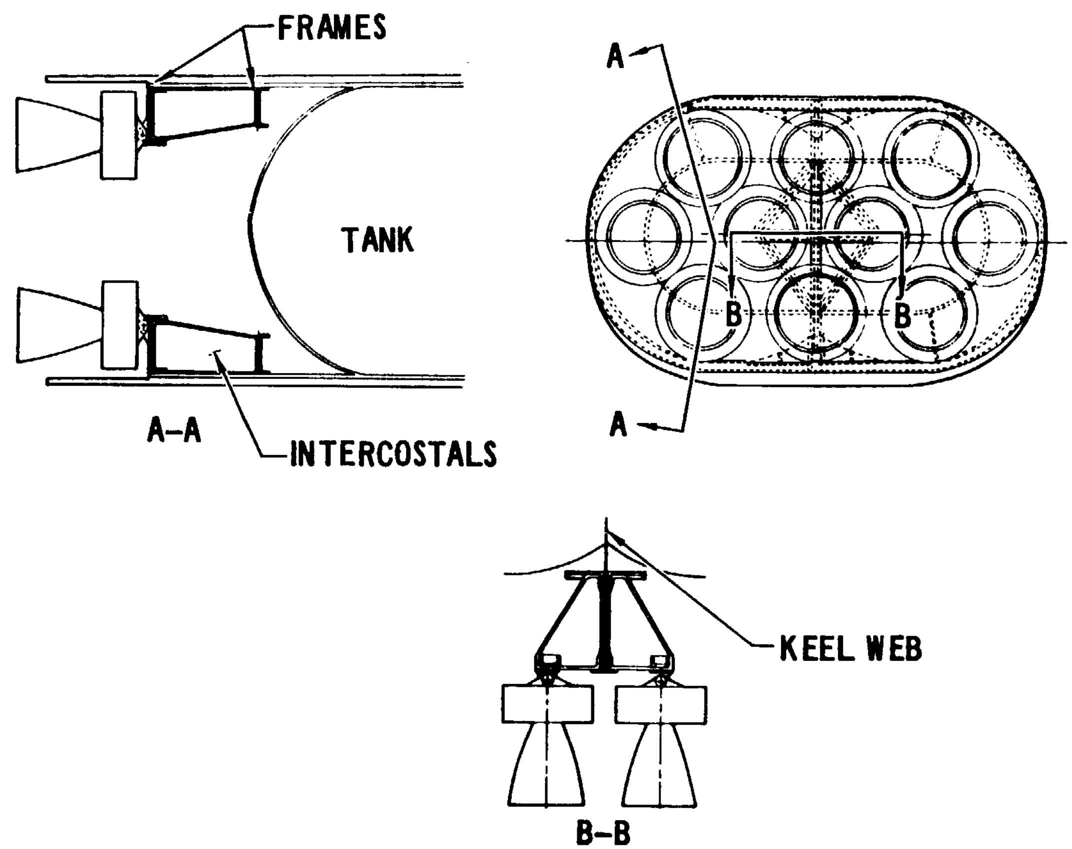
CARRIER THRUST STRUCTURE CONCEPT

Thrust structure consists of a simi-monocoque skirt with internal vertical web extended from the integral tank/fuselage shell, intercostals for local engine support and two major frames to support the intercostals. This structural arrangement leaves the center area open and easily accessible for installation of the propulsion system.

A design goal has been to introduce the engine thrust loads into the integral tank/fuselage structure as distributed loads, thereby, precluding the necessity for adding longerons for this loading condition. Engine loads are reacted locally by the inboard intercostal cap and the aft frame. Loads are sheared into the skirt with the resulting kick loads carried by the two major frames. Loads are then redistributed by the frames and skirt and introduced into the integral tank/fuselage structure as distributed loads. Structure designed for thrust loads provides a capability for pad support loads.

The forward frame is integrated with wing carry through structure. The frames also support the vertical tail and react carrier/orbiter attach loads.

CARRIER THRUST STRUCTURE CONCEPT



ILRVS-329F

CARRIER PROPULSION SYSTEMS

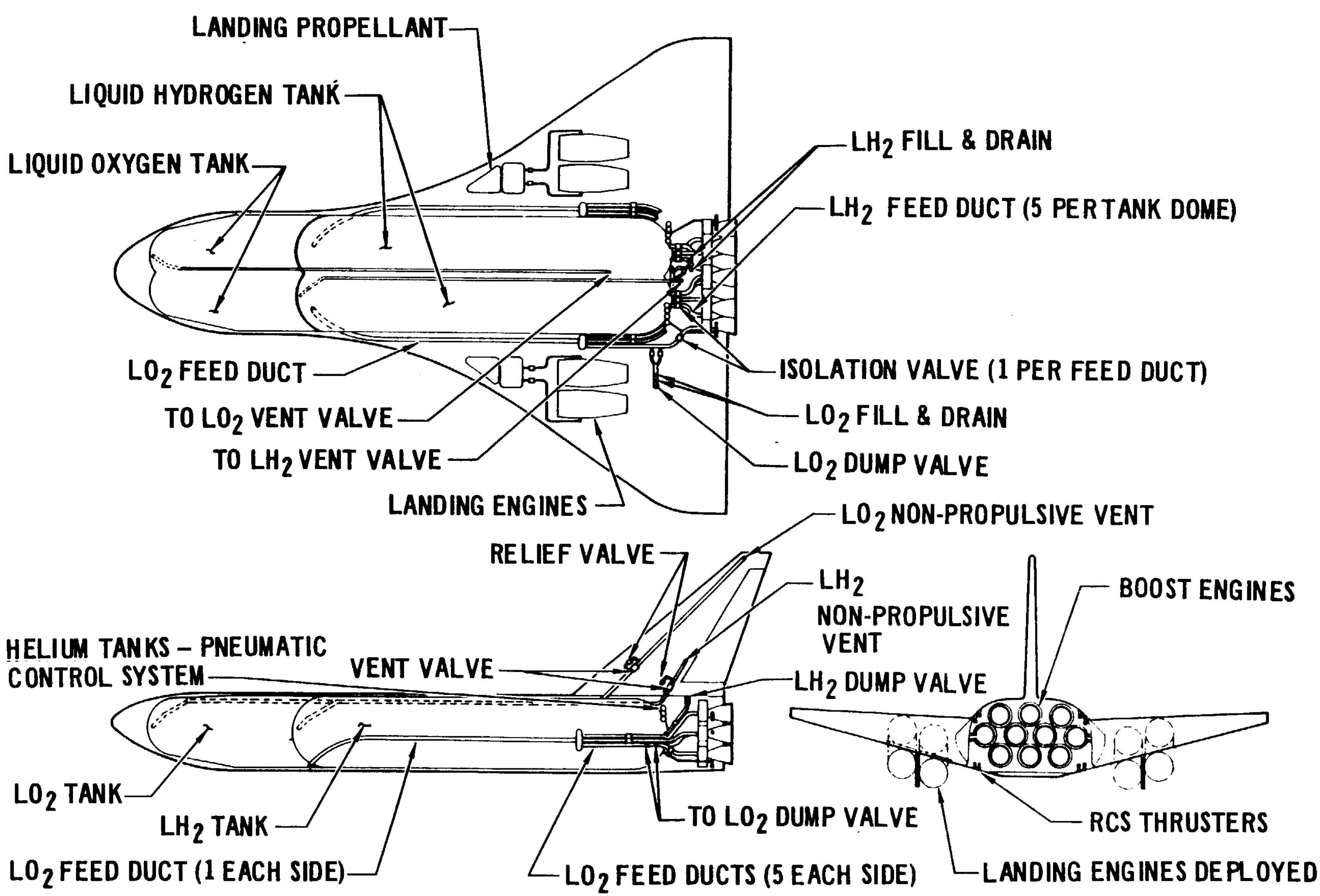
Propulsion systems for the carrier vehicle include boost, landing assist and an attitude control capability.

The boost propellant tanks are sized to provide a mixture ratio of 6 to 1. The oxidizer tank is forward of the hydrogen tank for favorable mass distribution and to minimize the engine gimbal requirements for thrust vector alignment through c.g. An oxygen feedduct is routed along either side of the vehicle, then manifolded to 5 of the boost engines. Separate hydrogen feedducts are routed from the tank to each engine. The oxygen tank is filled through the feed duct, and a separate duct is provided for hydrogen fill and drain with a disconnect near the base of the vehicle.

Four turbo-fan engines are provided for cruise and powered landing capability. They are stowed in the wing, and deployed below the wing for operation. JP-4 fuel is stowed in tanks forward of the engines. These tanks are insulated from heating during hypersonic flight.

An attitude control system provides vehicle orientation in the high altitude flight. Roll control is provided by coupled thrusters, and moments for pitch and yaw are provided by thrusters at the base of the vehicle. Hydrogen and oxygen for the attitude control system are stored as cryogenics in separate tanks.

CARRIER PROPULSION SYSTEMS



ILRVS-306F

CARRIER EQUIPMENT

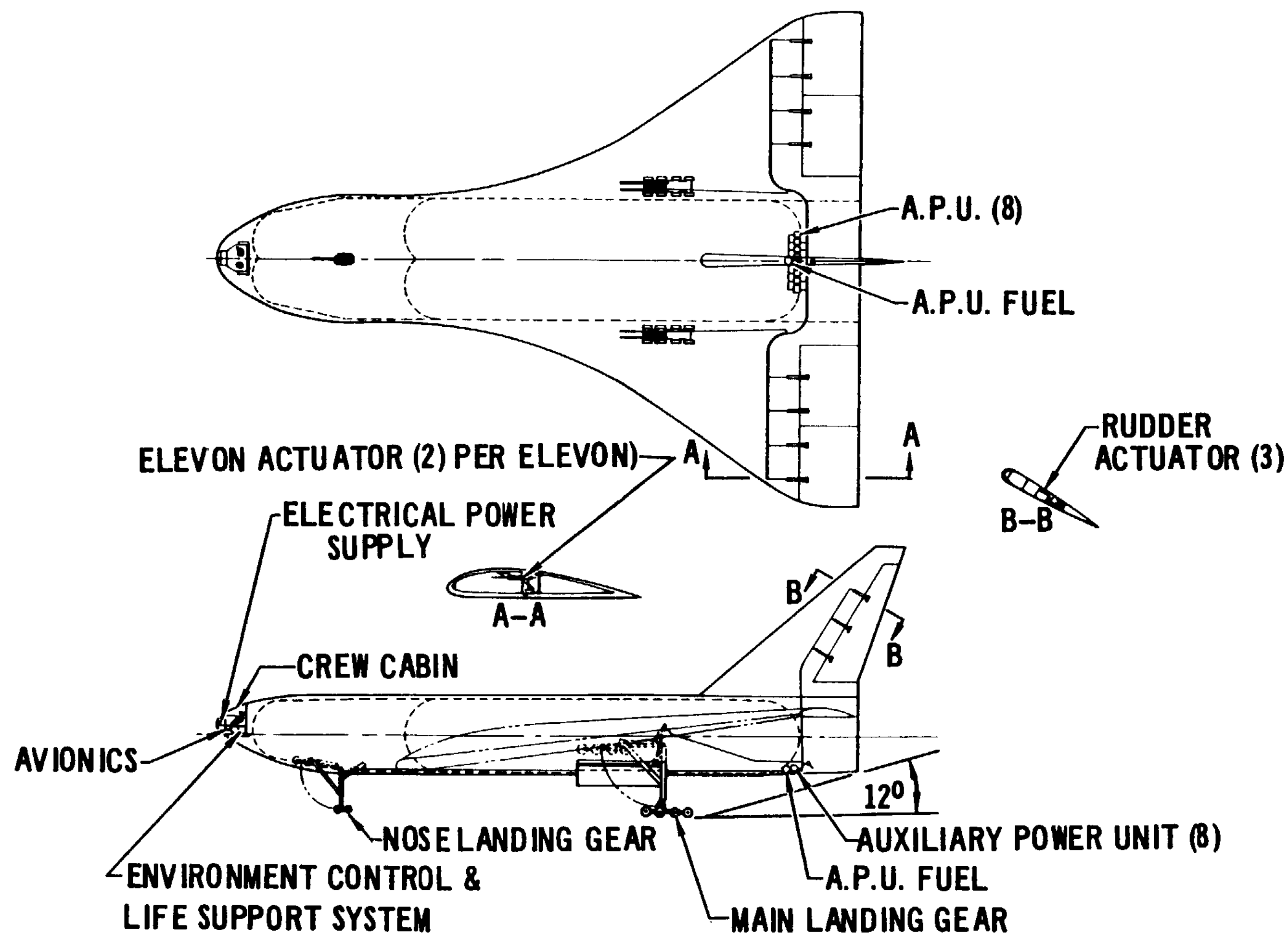
The equipment shown includes subsystems exclusive of structure and propulsion.

The crew cabin is sized for two men in a pressurized compartment. A shirtsleeve environment is provided by a single gas (O_2) environmental control system which occupies approximately 20 cu. ft. Avionics equipment requires 30 cu. ft. and the battery power supply requires 15 cu. ft. These systems are located at the forward end of the vehicle to assist in achieving vehicle balance.

Power for the aero control system actuators is supplied by 8 auxiliary power units. These are the same size units as those in the HL-10. They use hydrazine fuel, which is stored in an 8 cu. ft. sphere. These items are located aft of the main propellant tank. Three actuators are used for rudder rotation and a total of 8 for the elevons.

The main landing gear is slightly aft of landing c.g. with 8 tires required on each main gear assembly to limit landing load per tire. The forward gear is stowed in the cavity between the two oxygen tank lobes. Maximum angle of attack at landing, with the gear stroked, is 12°.

CARRIER EQUIPMENT



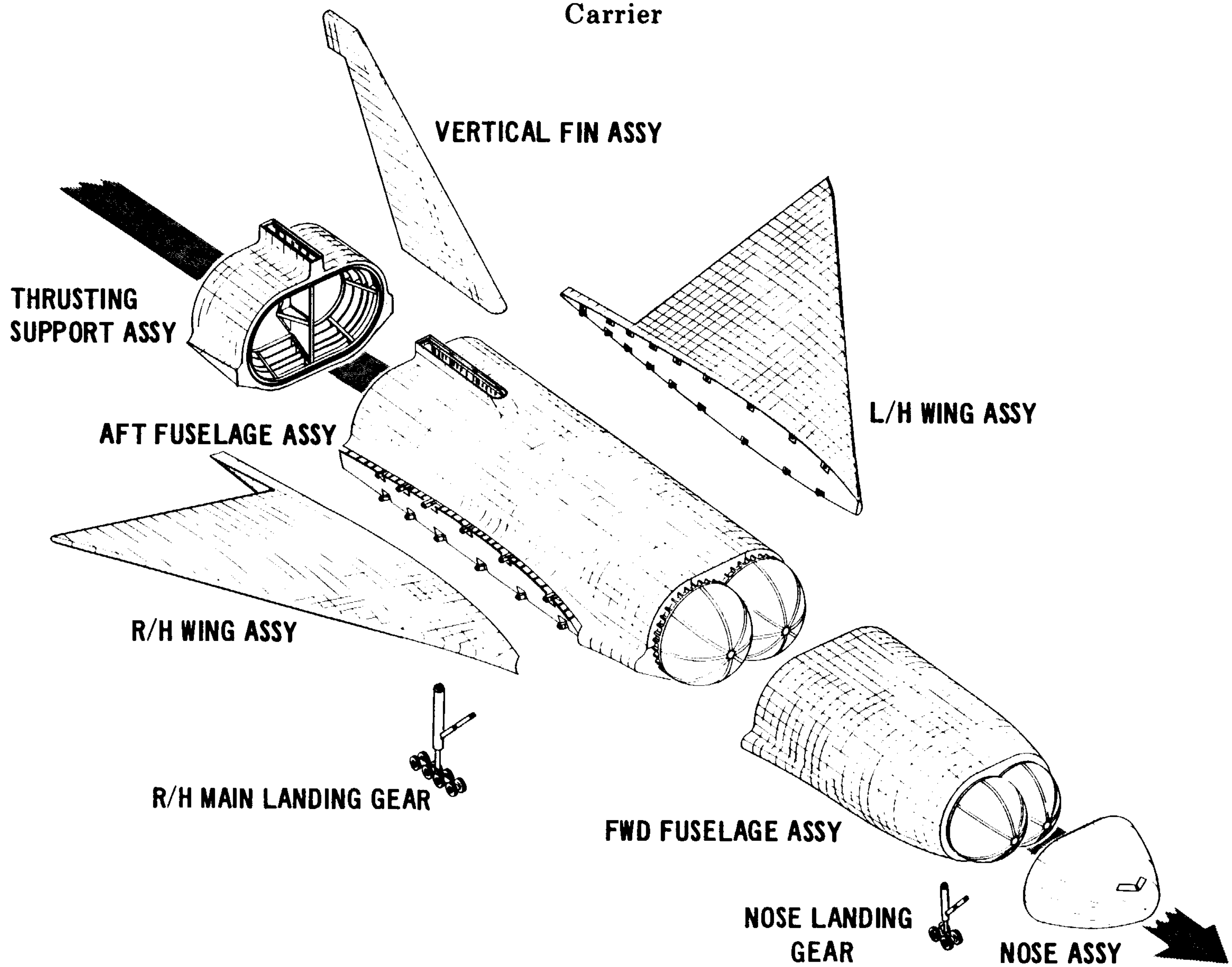
ILRVS-307F

PICTORIAL FLOW CHART - CARRIER

An orderly progression of the assembly activity for the clipped-delta carrier is shown in this chart. Major vehicle substructures shown in this exploded view provide information regarding their physical relationships and some orientation regarding assembly configurations. The forward and aft fuselage assemblies, which are integral tank structures, show the forward oxygen tanks and larger aft hydrogen tanks. The tanks are assembled, insulated and pressure tested separately. The thrust structure will house the ten boost engines.

PICTORIAL FLOW CHART

Carrier



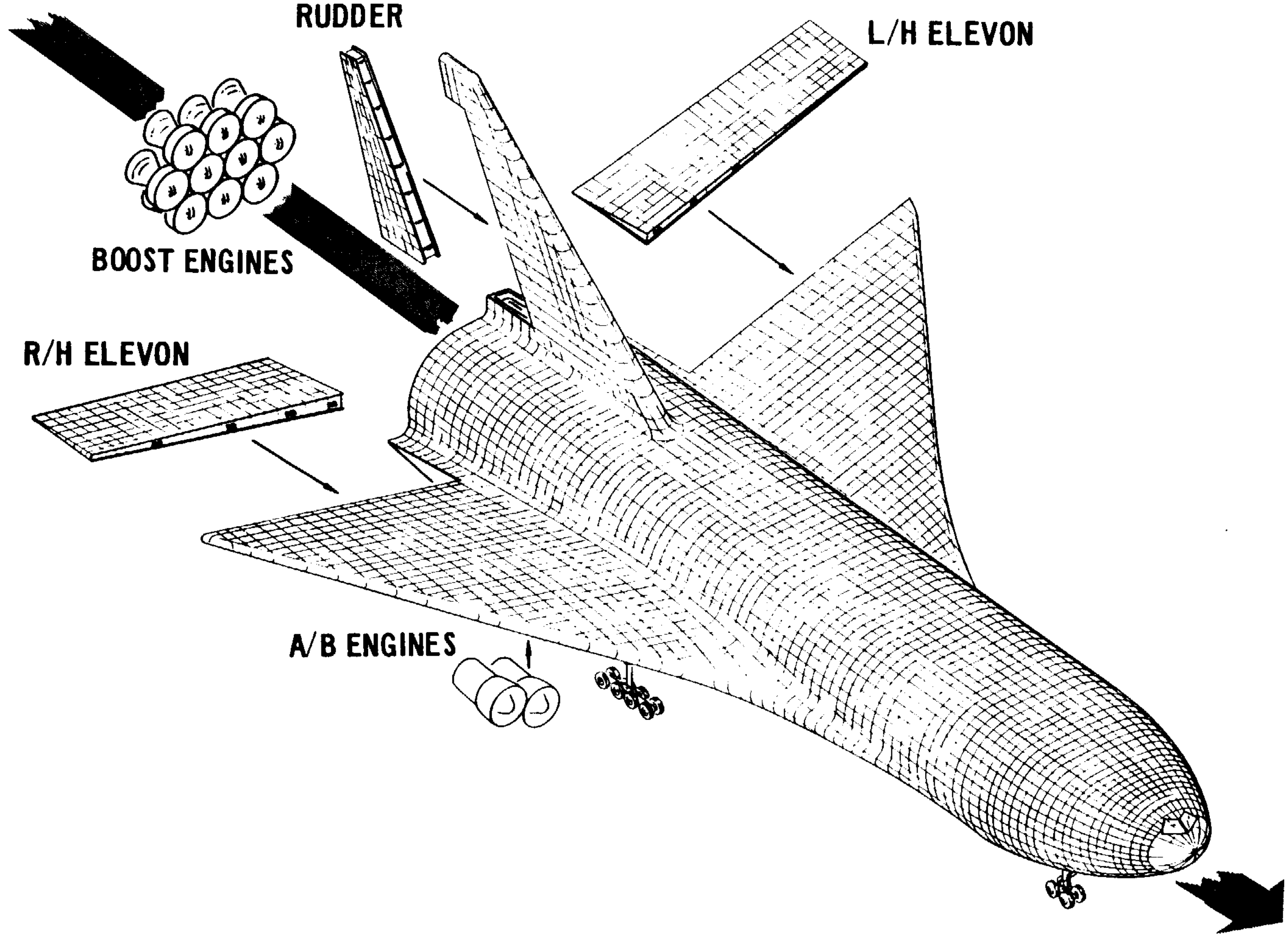
IL RVS 425F

PICTORIAL FLOW CHART - CARRIER

This chart shows the clipped-delta configuration with the fuselage structure assembled but showing the control surfaces, ten boost engines, jet-engines and the engine stowage compartment doors still to be assembled. The turbofan engines retract into the wing. Conventional multi-span structural arrangements are used in the wing and vertical tail.

PICTORIAL FLOW CHART

Carrier



IL RVS 424F

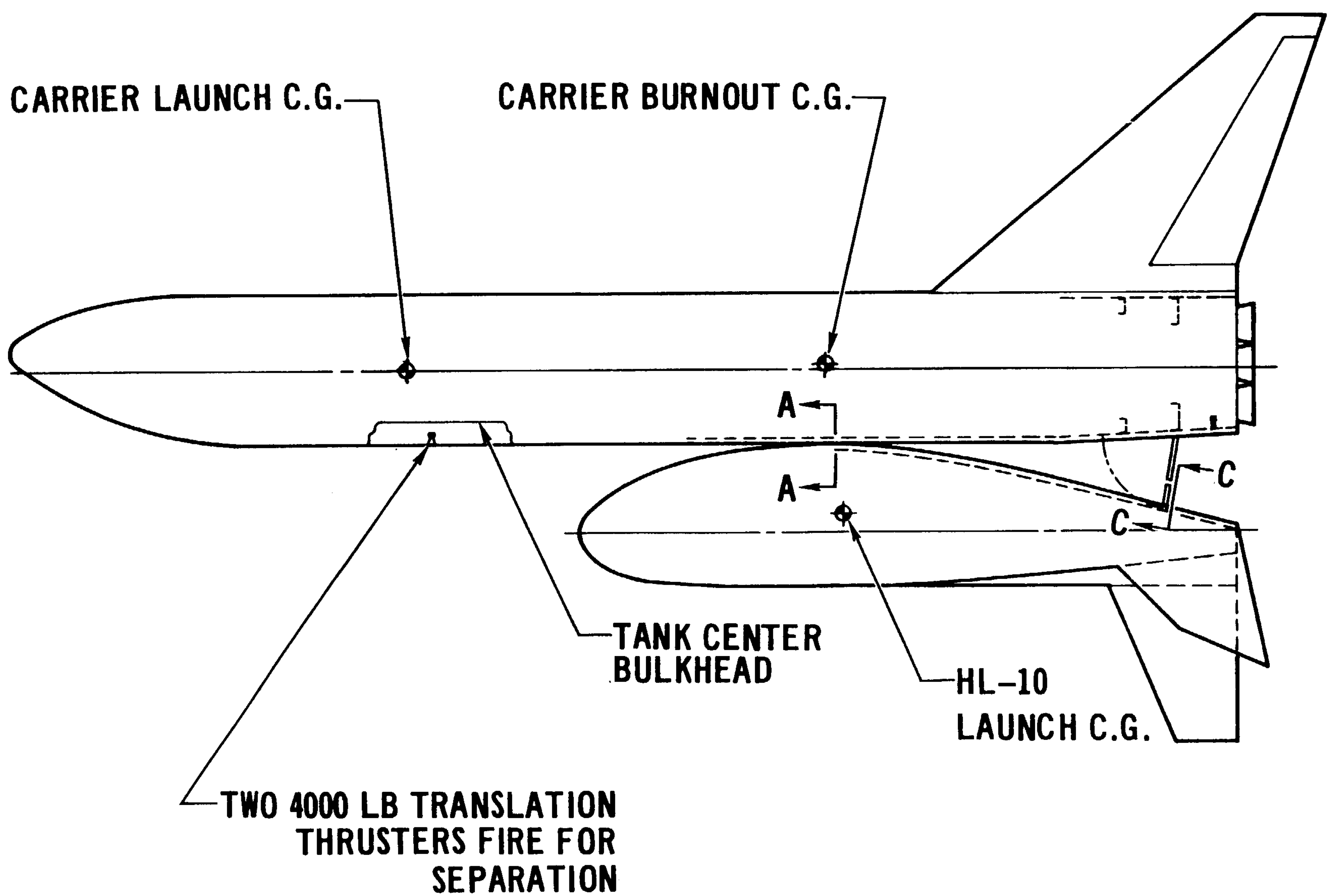
STRUCTURAL INTERFACE

The structural tie between carrier and orbiter consists of a three point attachment. Two fittings tie the vehicle lower surfaces together. These fittings are near the longitudinal c.g. of both vehicles at separation, and are located longitudinally and laterally to take maximum advantage of the existing body structure in both vehicles. They react longitudinal, vertical and lateral plus overturning moments in the lateral and longitudinal planes.

A separate support arm is provided near the aft end of the vehicles to complete the structural interface. This arm is allowed to take loads long it's axis, but offers no resistance to side loads and vehicle axial loads. It therefore combines with the forward fittings to react overturning moments in the vertical plane. This fitting is retracted into the carrier after separation and the thermal protection on it restores moldline smoothness in the vehicle aft lower surface.

Translation thrusters on the carrier provide separation of the vehicles. The thrust is provided by using a combination of pitch down thruster and a separate thruster forward of the carrier c.g. The carrier is thus translated, and pitched up away from the orbiter.

STRUCTURAL INTERFACE



ILRVS-305F-A

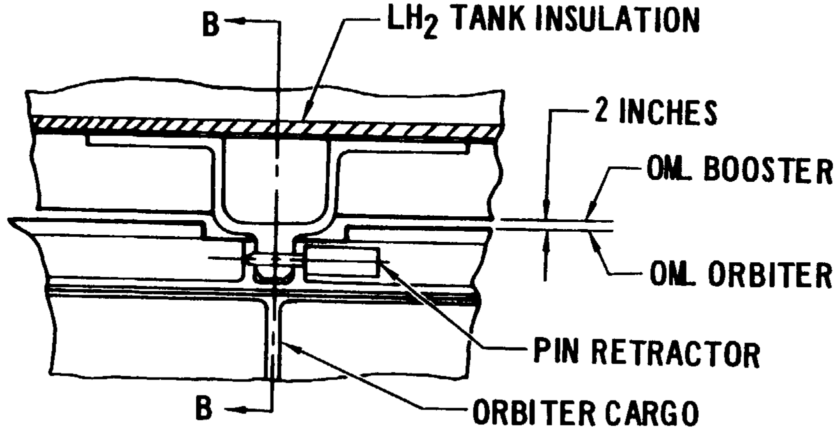
STRUCTURAL INTERFACE (CONTINUED)

Details of the structural interface are shown in these views. Views A-A and B-B are typical of the two forward fittings. The carrier structure extends past moldline into a cavity in the orbiter where it is fastened to the orbiter by a retractable pin. The orbiter fittings are aligned with the intersection of the lateral bulkhead forward of the cargo bay and the longitudinal webs on either side of the cargo bay. The attachment between the interfacing fittings is located close to the orbiter structural skin to minimize induced moments. This arrangement minimizes the structural penalty to the orbiter for providing launch load capability. Doors on the orbiter lower surface close after separation to maintain a smooth moldline surface.

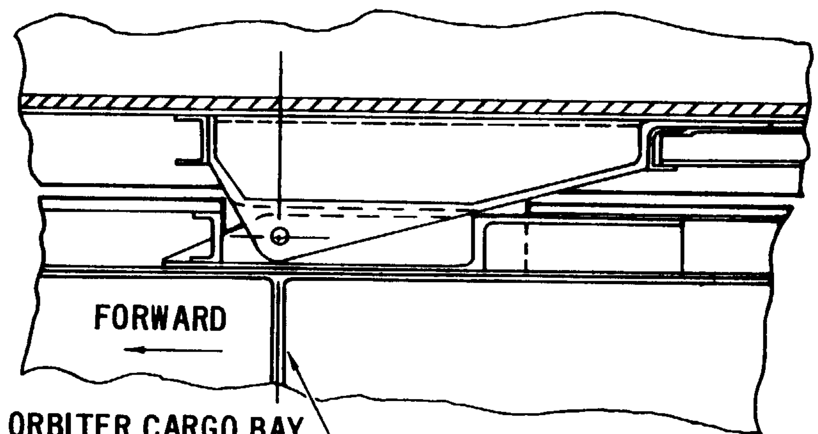
The carrier fittings transfer loads to longerons and to the structural skin. The overturning moment induced by the offset load is reacted by existing body rings. This fitting is permitted to remain outside of moldline during return and has the capability of withstanding the heating caused by the surface discontinuity.

The aft fitting is structurally attached as shown in view C-C. This fitting also extends into a cavity on the orbiter lower surface. The two vehicles are tied together by a retractable pin like the forward fittings. Loads are carried in the orbiter by the forward thrust structure bulkhead. They are reacted in the carrier by the aft thrust structure ring. The orbiter lower surface contour at this point is alos maintained by a cover which closes after separation.

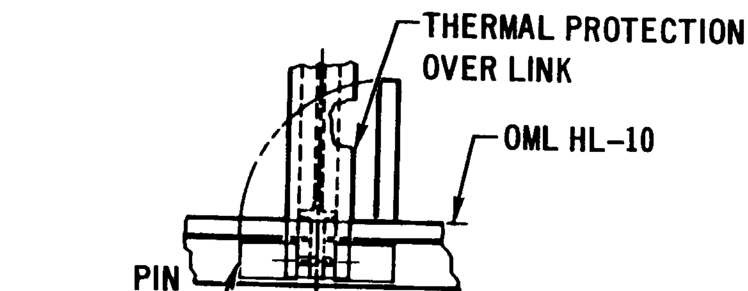
STRUCTURAL INTERFACE CONT'D



A–A
TYP–2 PLACES



B–B



C–C

ILRVS–305F–B

SEPARATION TRAJECTORY HL-10 WITH RESPECT TO CARRIER

A digital computer program was developed and utilized to generate separation trajectories. The basic assumptions are:

1. All quantities perform in a unique describable manner
2. Aerodynamics negligible

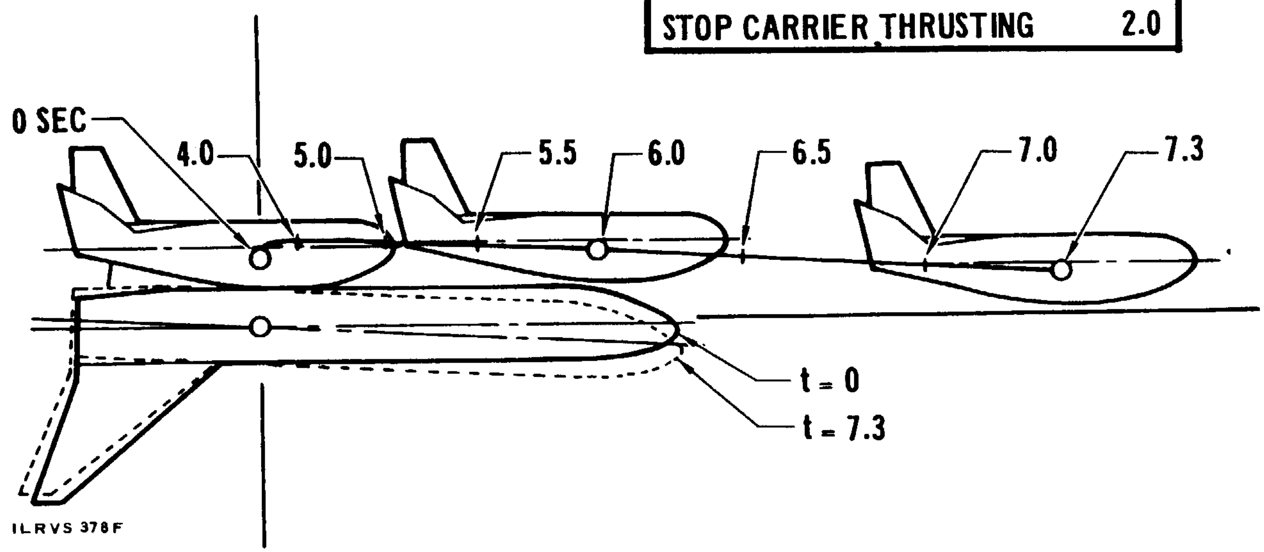
The sequence of events prior to separation are:

1. The combined vehicles have flight path and body rates nulled
2. The carrier thrust decays to zero
3. Rate damping continues up to separation
4. After carrier thrust is zero, a delay of .1 seconds occurs before separation
5. The connections between the two vehicles are removed without disturbing either vehicle

SEPARATION TRAJECTORY

TIME OF PERTINENT EVENTS – SECONDS

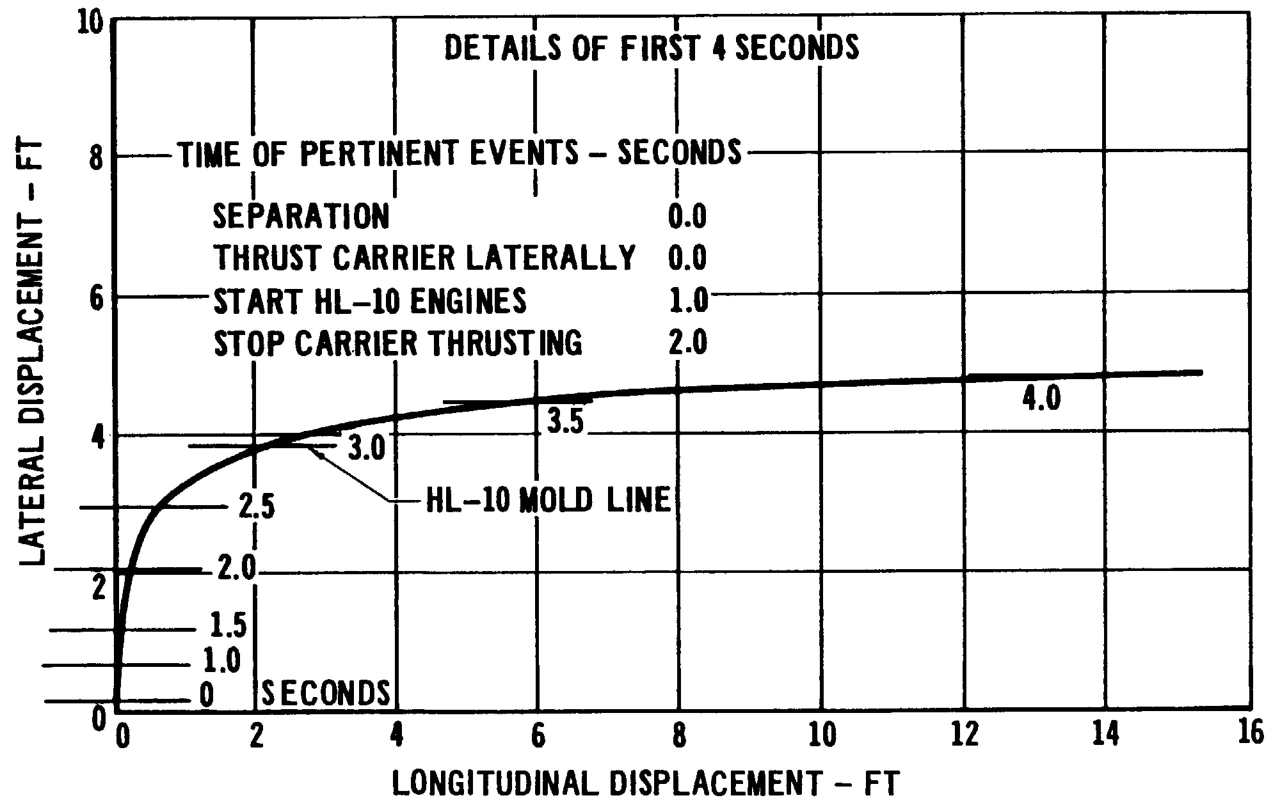
SEPARATION	0.0
THRUST CARRIER LATERALLY	0.0
START HL-10 ENGINES	1.0
STOP CARRIER THRUSTING	2.0



MOLDLINE SEPARATION HL-10 WITH RESPECT TO CARRIER

While a satisfactory separation using this technique has been demonstrated, the study was not exhaustive. Further work to examine all significant contributions in detail remains to firmly establish this technique. Among these items is that of HL-10 engines plume impingement on the carrier. Since the minimum lateral clearance between the HL-10 engines and the surface of the carrier is several nozzle diameters, no effect is anticipated which would invalidate this separation technique, but the effects such as pressures, temperatures, and contamination remain to be analyzed.

MOLD LINE SEPARATION



ILRVS-379F

DETAILED WEIGHT SUMMARY - HL-10

The major part of the primary structure consists of integral tankage. This structure is an aluminum ring stiffened shell with longitudinal stiffeners. Due to the unconventional shape, biaxially loaded tension webs have been added. Also included in the structural weights are the tank baffles, bulkheads and insulation. The average unti weight for this structural concept is 4.70 lbs/ft^2 of vehicle outer moldline wetted area.

The thermal protection system weights consist of radiative panels. The weights shown reflect materials selected comparible with high reuse capability and reentry temperatures. Beaded titanium shingles are used over 61% of the top of the vehicle. The remaining top, entire sides and approximately 10% of the bottom utilize René shingles. TD-NiCr shingles cover the remaining 90% of the bottom and leading edges. Columbium shingles cover the base of the vehicle as well as the nose cap. A passive insulation system was chosen for the balance of the thermal protection system. The necessary insulation, back-up and installation weights average .90 lbs/ft^2. The average weight for the thermal protection system described above is 2.18 lb/ft^2 of vehicle outer moldline wetted area.

The aero surface weights include lower fixed fins, elevons, and a vertical tail. Components considered in their weight estimates include the shell weight, leading edge weight, hinge weight, actuator weight and thermal protection weight. The average for the primary structure (excluding thermal protection) of the tail elements is 2.94 lbs/ft^2 for the movable surfaces and 6.48 lb/ft^2 for the fixed surfaces. The above unit weights are based on the projected area of the aero surfaces.

The other main sources of weight are to be found in the propulsion system weight elements. These elements include the Boost Propulsion System, the Orbit Propulsion System, the Entry Control System and the Landing Propulsion System. Included in the Boost Propulsion System is the thrust structure which is made up of a network of aluminum trusses whose weight is .003 (total vac. thrust). The weight allocated for pressurization and residuals is .005 (usable prop.). The Orbit Propulsion System is made up of 7% inert weight and 93% usable propellant weight and was sized for an on orbit ΔV of 2000. The Entry Control and Landing Propulsion Systems inert weights are 46 and 57% respectively. A total of 544,340 lbs. of usable propellant is loaded which represents 74.5% of the vehicle gross weight.

The landing gear weight estimates utilized a correlation of historical airplane data. This gear is a conventional rolling gear whose weight approaches 4.5% of the landing weight. Personnel and their provisions and the environmental control system were sized for two men and a mission duration of seven days. All other subsystems such as integrated avionics, prime power and the separation provisions are typical of systems needed to meet the requirements of a logistic mission.

DETAILED WEIGHT SUMMARY

HL-10

ITEM	WEIGHT	ITEM	WEIGHT
AERO SURFACES	(9,150)	INTEGRATED AVIONICS	2,200
ELEVONS	2,060	CREW AND FURNISHINGS	600
SIDE FINS	5,320	ENVIRONMENTAL CONTROL	1,940
VERTICAL TAIL	1,770	BOOST PROPULSION SYSTEM	(525,620)
BODY STRUCTURE	(46,030)	INERT & TRAPPED PROP	25,630
SKIN	30,360	PROPELLANT	499,990
FRAMES	7,280	SECONDARY PROP SYSTEM	(38,480)
BULKHEADS	4,770	INERT & TRAPPED PROP	4,080
TANK INSULATION	3,620	MANEUVER PROPELLANT	32,450
THERMAL PROTECTION	(27,970)	ATTITUDE CONTROL PROP	1,930
BODY	20,900	LANDING PROPULSION SYSTEM	(23,490)
ELEVONS	2,100	INERT & TRAPPED PROP	13,520
SIDE FINS	3,160	PROPELLANT	9,970
VERTICAL TAIL	1,810	CONTINGENCY	13,990
LANDING SYSTEM	8,360	BALLAST	0
ELECTRICAL POWER	3,860		
AERODYNAMIC CONTROLS	1,620	GROSS PAD WEIGHT	730,220
AERO APU SYSTEM	900		
HYDRAULIC SYSTEM	830		
RANGE SAFETY & ORDNANCE	200		
CARGO & SUPPORTS	25,000		

DETAILED WEIGHT SUMMARY
(CARRIER)

A weight summary similar to the HL-10 is presented for the carrier. The empennage weights for the carrier include the vertical fin and the fixed wings. The weight estimates used on the vertical fin were similar to those used on the HL-10 tail and its unit weights is 2.68 lbs/ft^2 of wetted area. The wing weights include the basic shell weight, plus the bending material such as spars, stringers, etc. plus the enclosure weight. Weight estimates for the wing were derived using a conventional aircraft wing estimation technique and its unit weight is 6.12 lb/ft^2 of wetted area.

The primary structure of the carrier is similar to that of the HL-10 and it is only necessary to note that this weight includes the primary structure, the tank bulkheads, and baffles and the tank insulation. The average weight of the carrier structural design is 5.66 lb/ft^2 of body outer mole line wetted area.

The thermal protection system consist of radiative panels. The upper 1/2 of the vehicle consists of titanium panels of which 66% are beaded while the remaining 33% are corrugated. The lower 1/2 of the vehicle is comprised almost solely of titanium corrugated panels. The only exception is the Rene panels in the nose area. The base heat protection is made up of columbium panels. The average back up, installation and insulation weight penalty is .52 lbs/ft^2. The average weight for the thermal protection system fo for the body is .98 lb/ft^2. Thermal protection of the wing and tail surfaces is not required. However, on the wing in the areas of the landing propellant tanks and engines and landing gear insulation was provided. This insulation is a 1 inch layer of micro quartz and its average weight is .5 lbs/ft^2.

All other subsystems are the same as the HL-10 except for the personnel provisions and the environmental control system which have been sized for two men and a mission duration of two days. As in the HL-10, no ballast is required to place the center of gravity in the required position during the entry maneuver.

DETAILED WEIGHT SUMMARY CARRIER

ITEM	WEIGHT	ITEM	WEIGHT
AERO SURFACES	(90,290)	INTEGRATED AVIONICS	1,570
TAIL	6,860	CREW AND FURNISHINGS	600
WING	83,430	ENVIRONMENTAL CONTROL	430
BODY STRUCTURE	(112,990)	BOOST PROPULSION SYSTEM	(2,292,870)
SKIN	51,100	INERT & TRAPPED PROP	115,750
FRAMES	13,580	PROPELLANT	2,177,120
BULKHEADS	15,820	SECONDARY PROP SYSTEM	2,050
TANK INSULATION	7,230	LANDING PROPULSION SYSTEM	(97,750)
SUPPORT	25,260	INERT & TRAPPED PROP	36,810
THERMAL PROTECTION	(23,630)	PROPELLANT	60,940
BODY	19,630	RANGE SAFETY & ORDNANCE	200
TAIL	0	BALLAST	0
WING	4,000	CONTINGENCY	39,060
LANDING SYSTEM	20,290		
ELECTRICAL POWER	2,430	GROSS PAD WEIGHT	2,688,990
AERODYNAMIC CONTROLS	2,180		
AERO APU SYSTEM	1,400	GROSS LIFT OFF WEIGHT	2,671,920
HYDRAULIC SYSTEM	1,250		

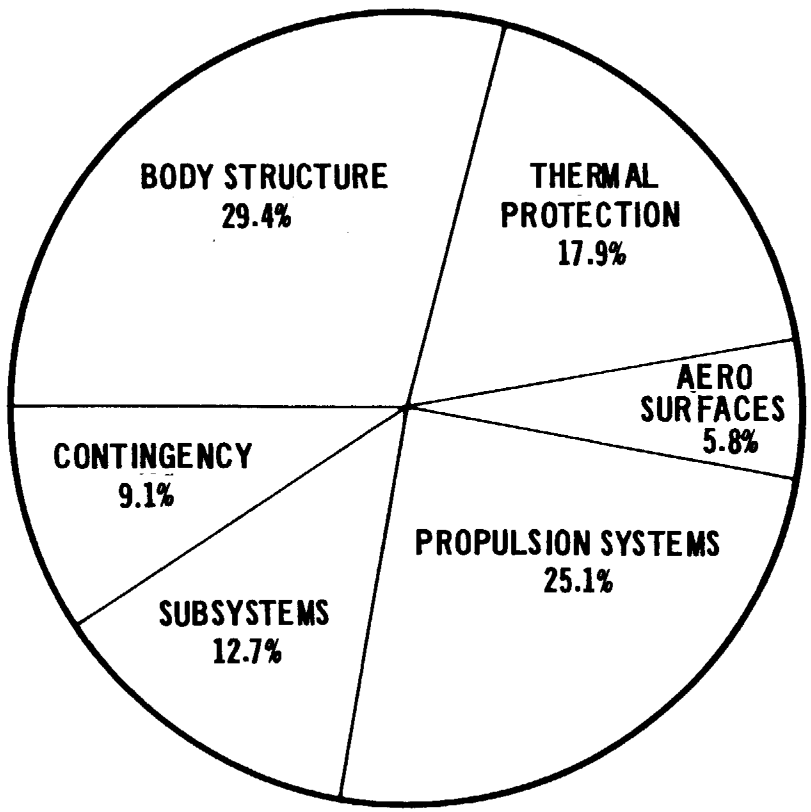
ILRVS-340F

DRY WEIGHT DISTRIBUTION

The pie charts present in clear and simple form the major weight distribution of the carrier and orbiter in the dry weight condition. It can be quickly determined which groupings contribute most to the weight makeup. This information can be used to help determine which areas to concentrate efforts at weight reduction, cost reduction, etc. for maximum effect.

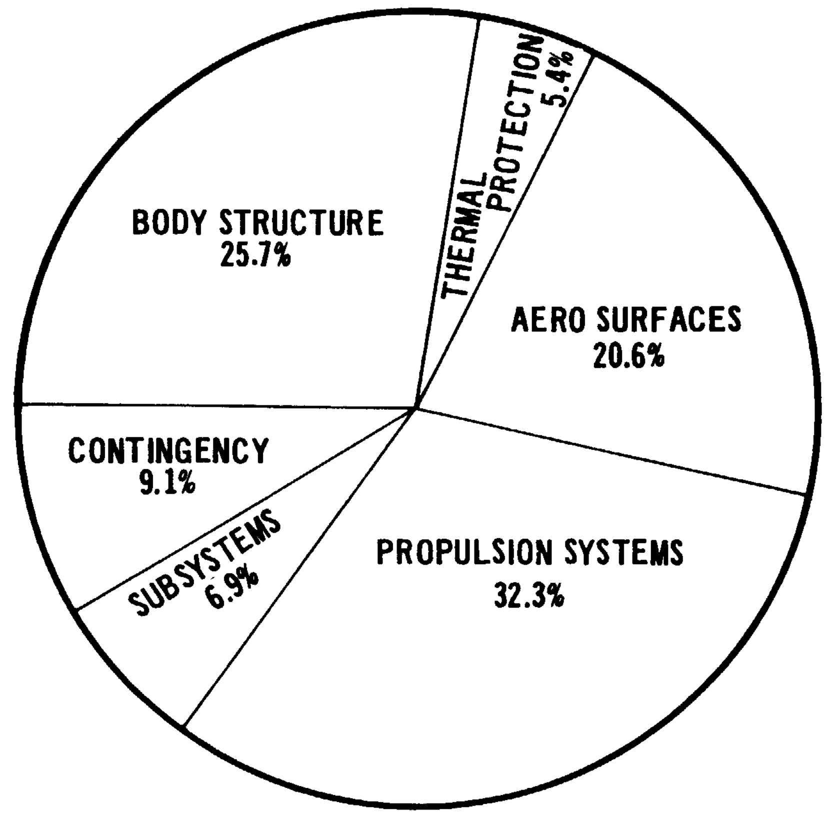
DRY WEIGHT DISTRIBUTION

ORBITER



DRY WEIGHT = 156,470 LB

CARRIER



DRY WEIGHT = 438,230 LB

ILRVS-342F

CARRIER TEST SUMMARY

A series of exploratory wind tunnel tests were performed with the Clipped Delta Wing Carrier configuration at the NASA Langley Research Center. The tests included the Carrier configuration alone as well as the 2-stage ascent configuration, which consisted the first stage carrier attached to the HL-10 orbiter. Force and moment tests were conducted at a subsonic mach number of 0.3 and a hypersonic mach number of 10.4. Thermographic tests on the carrier alone were conducted at Mach number 10.4, using phase change material to indicate first order heating effects. The carrier model was an intermediate configuration and did not exactly match the final study configuration; however, the lines were sufficiently close to insure that indicated trends are good representations of the vehicle characteristics. The aerodynamics forces for both the carrier configuration and the ascent phase configuration have been normalized with the carrier vehicle theoretical wing area and the moments were normalized with the wing area and the corresponding mean aerodynamic chord. In the figures which present the carrier along longitudinal moments, the moment reference point is at the 66% station on the vehicle centerline. The total planform area of the carrier vehicle is 26% greater than the theoretical wing area and the mean aerodynamic chord is 44% of the body length. A control deflection convention was adopted which defined negative deflection as trailing edge up.

The summary table on the facing page lists the ranges of the pertinent test variables which were obtained at the two mach number conditions.

CARRIER TEST SUMMARY

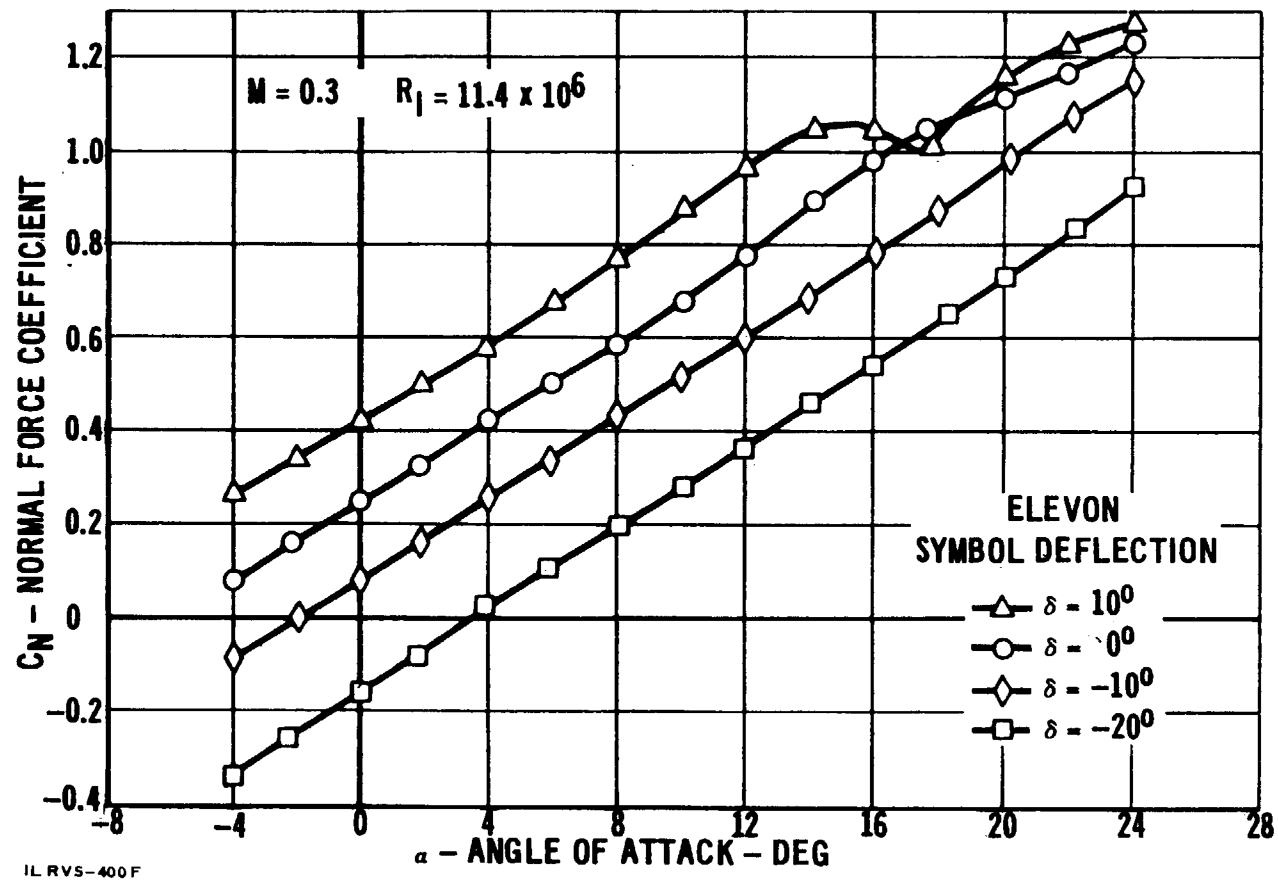
<table>
<tr><th>LRC LOW TURBULENCE PRESSURE TUNNEL
(M = 0.30)</th><th>LRC CONTINUOUS FLOW HYPERSONIC TUNNEL
(M = 10.4)</th></tr>
<tr><td>FORCE TESTS
● CARRIER
$-7^0 < \alpha < 24^0$
$-20^0 < \delta_e < +10^0$
$-5^0 < \beta < +0^0$
$\frac{t}{c} = 0.09, 0.15$
HIGH AND LOW WING
WING FAIRING OFF AND ON
● ASCENT CONFIGURATION
$-7^0 < \alpha < 24^0$
$-5^0 < \beta < 5^0$</td><td>FORCE TESTS
● CARRIER
$-2^0 < \alpha < 28^0$
ELEVON ON AND OFF
● ASCENT CONFIGURATION
$-4^0 < \alpha < 10^0$
HEAT TRANSFER
● CARRIER
$M = 10.4$
$\alpha = 15^0$ AND 50^0
HIGH AND LOW WING
FAIRING OFF AND ON</td></tr>
</table>

ILRVS–358F

NORMAL FORCE COEFFICIENT

The adjacent page shows the tested subsonic normal force characteristics of the Clipped Delta Wing carrier. The model wing was constracted with a NACA 4415 airfoil section, which provided linear normal force variations up to approximately 14° angle of attack. Beyond this point, the test data obtained with positive elevon deflection exhibited wing stall characteristics. However, the data with the negative elevon deflections, which are presently being used for trim, did not show any stall indications over the angle-of-attack range included in the test. As indicated in the figure, a sufficient amount of elevon effectiveness is available for control. This effectiveness is due mainly to the size of the elevons since the combined plan area of both panels is approximately 18% of the theoretical wing area. The vehicle reference area is the theoretical wing area.

CLIPPED DELTA WING CARRIER SUBSONIC TEST DATA

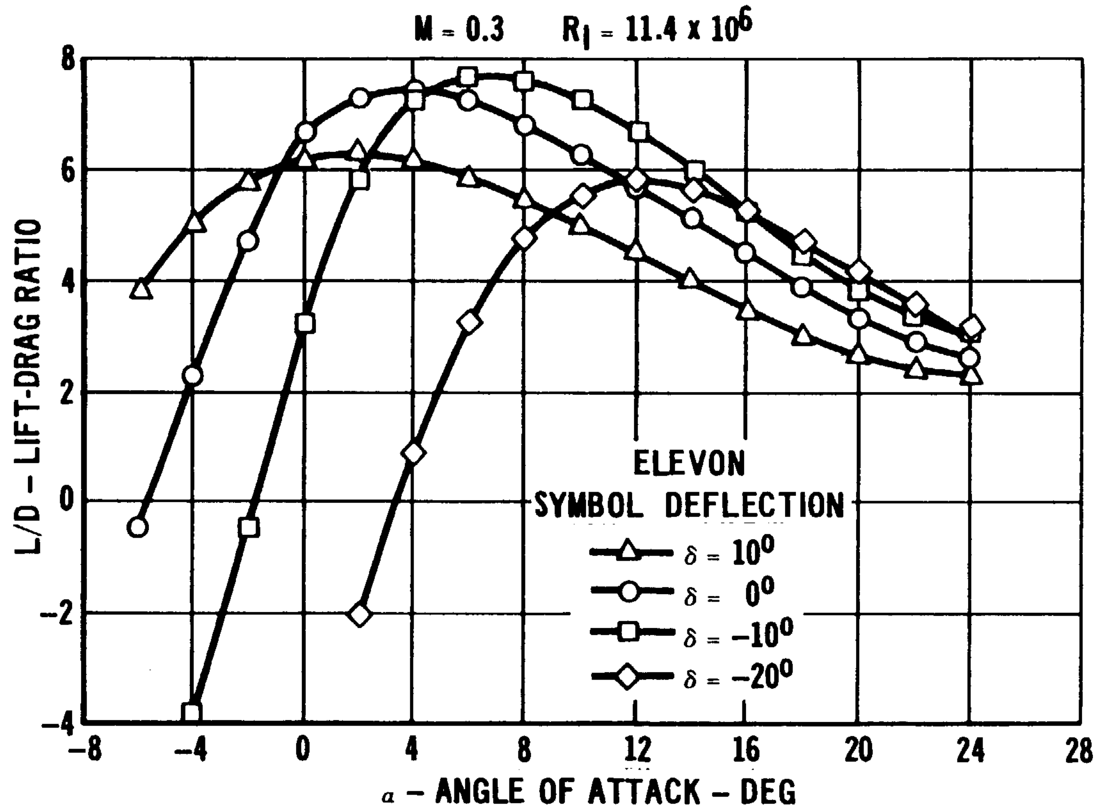


IL RVS-400F

SUBSONIC UNTRIMMED LIFT-DRAG RATIOS

The subsonic lift-drag ratios for the untrimmed clipped delta wing carrier are shown on the facing page. These data are shown for each of the control deflections tested, and the maximum value of 7.65 is obtained with a negative 10° deflection at 7° angle of attack. Elevon sign convention defines a negative deflection as tracking edge up. A cross plot of the test data indicates that the maximum obtainable value of L/D is 7.72; however, this is an ideal value, since the vehicle would have to trim in an unstable condition with the angle of attack and control deflection necessary to obtain this maximum.

CLIPPED DELTA WING CARRIER SUBSONIC TEST DATA

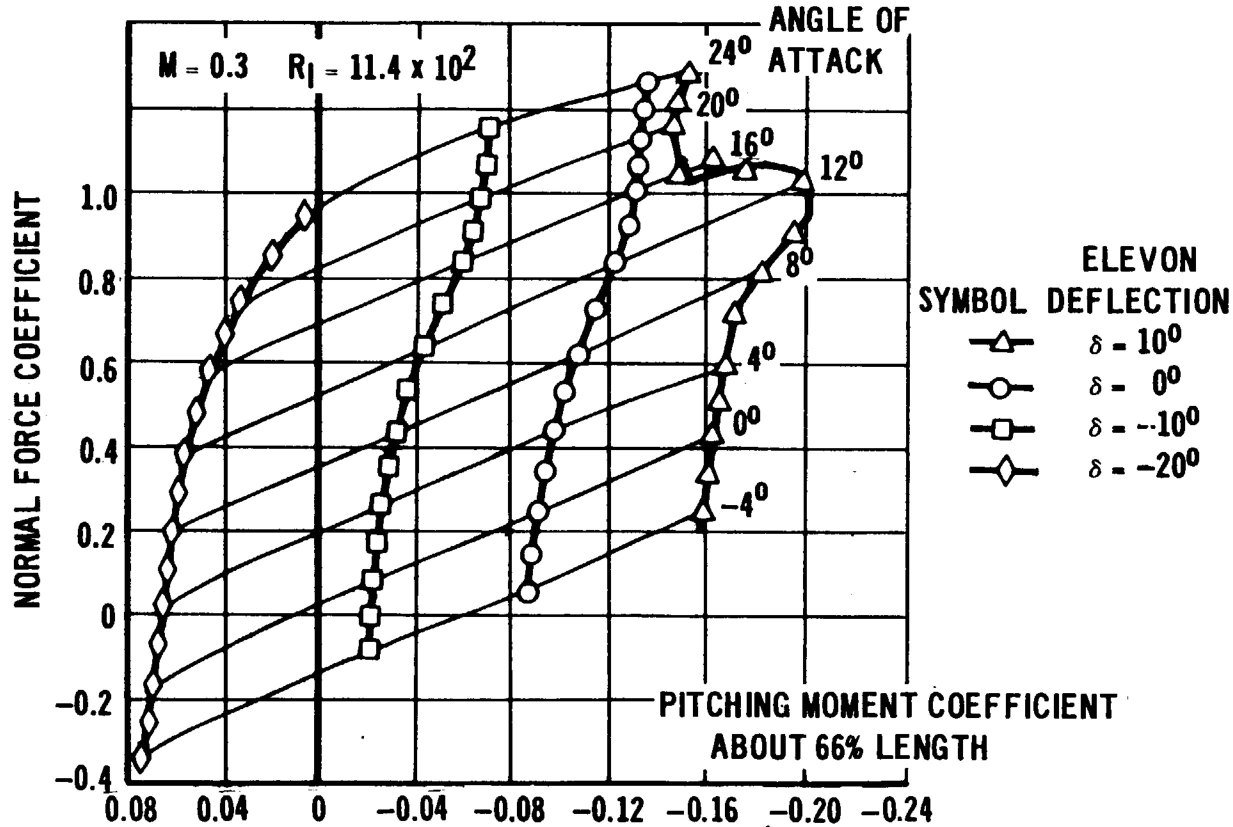


ILRVS-402F

PITCHING MOMENT COEFFICIENT

The subsonic test results which defined the stability characteristics of the Clipped Delta Wing Carrier are shown on the following page. The moment reference length is the mean aerodynamic chord of the theoretical wing plan; and is 44% the vehicle length. The moment reference point is at 66% of the vehicle length. As the figure indicates, the vehicle is stable over the entire trim angle of attack range shown, but exhibits undesirable moment characteristics in the untrimmed region where wing stall is suspected. Although a large C_{m_o} exists for the $\delta = 0°$ configuration, the elevon control effectiveness is sufficient to provide trim up to at least 24 degrees angle of attack, and the data trends indicate that larger deflections would yield even higher stable trim angles.

CLIPPED DELTA WING CARRIER SUBSONIC TEST DATA

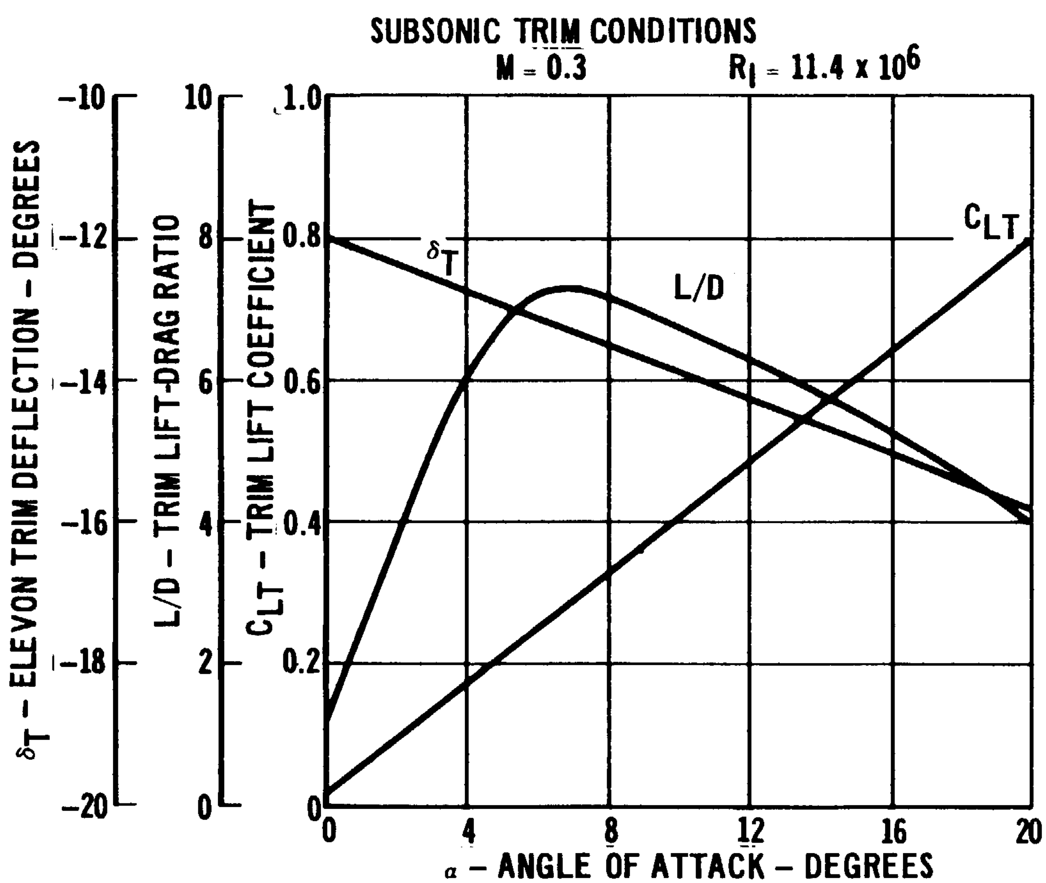


ILRVS-401F

SUBSONIC TRIM CONDITIONS

The subsonic trip characteristics of the Clipped Delta Wing Carrier are summarized in the adjacent figure. The trim reference point is at the 66% station on the vehicle centerline. The maximum value of trim L/D is 7.4, which occurs at 7.5° angle-of-attack with a negative 13.4° elevon deflection. The vehicle has a sufficient amount of trim control authority, since one degree of control deflection can increment the trim angle of attack by 5.5°. The trim characteristics were obtained by working directly with the test data, and at present no corrections have been made to account for full scale effects. The vehicle reference area is the theoretical wing area.

CLIPPED DELTA WING CARRIER



ILRVS-403F

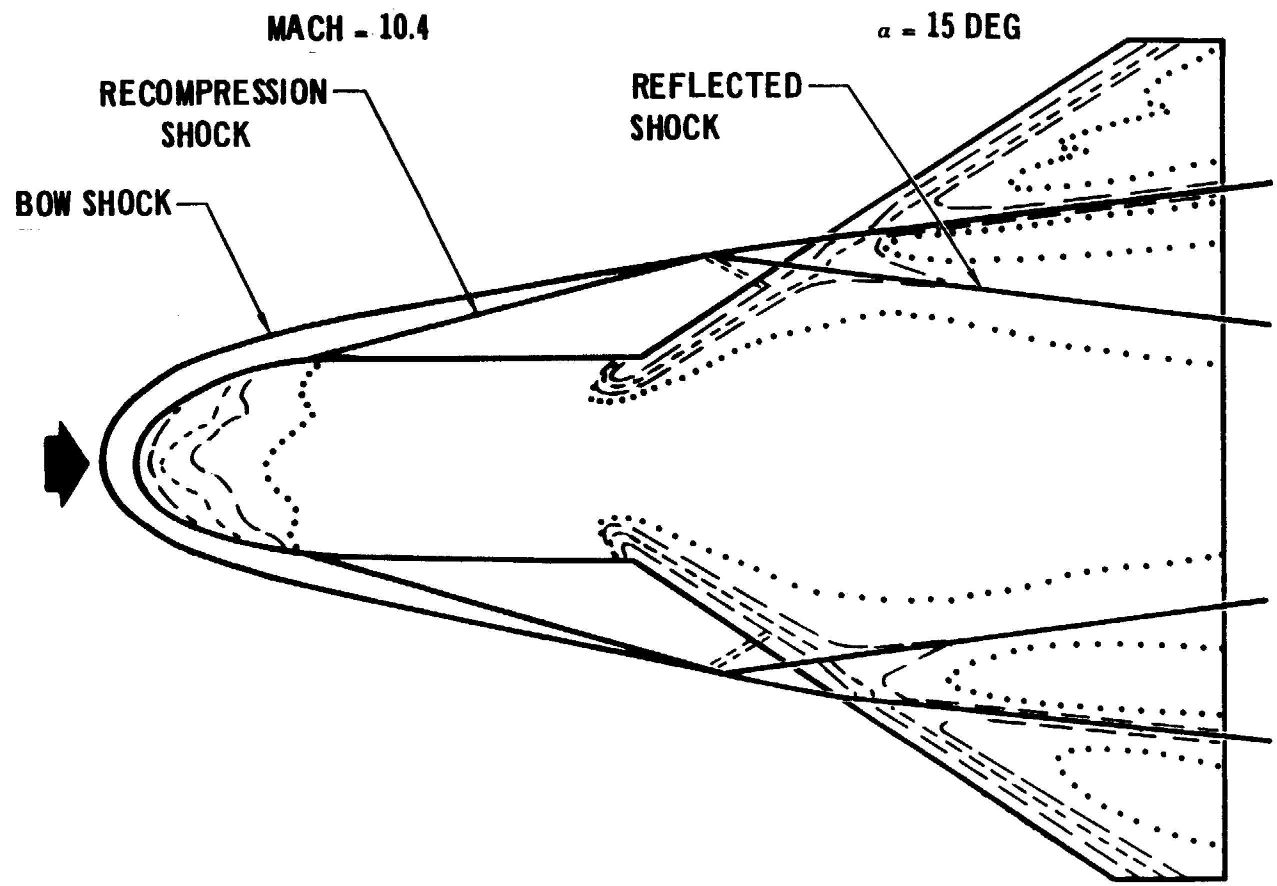
SHOCK PATTERN

Experimental heat transfer test results indicate regions of increased heating rates on the wing planform at a 15° angle of attack. These regions are probably a result of shock interaction.

The main bow shock is followed by expansion and recompression shock. Intersection of the recompression shock with the main bow shock results in reflection of the weaker shock and intersection of both shock waves with the wing leading edge shock.

As a result of these interactions, the estimated hypersonic aerodynamics differ significantly from the test data since the interactions cannot be accounted for analytically.

SHOCK PATTERN

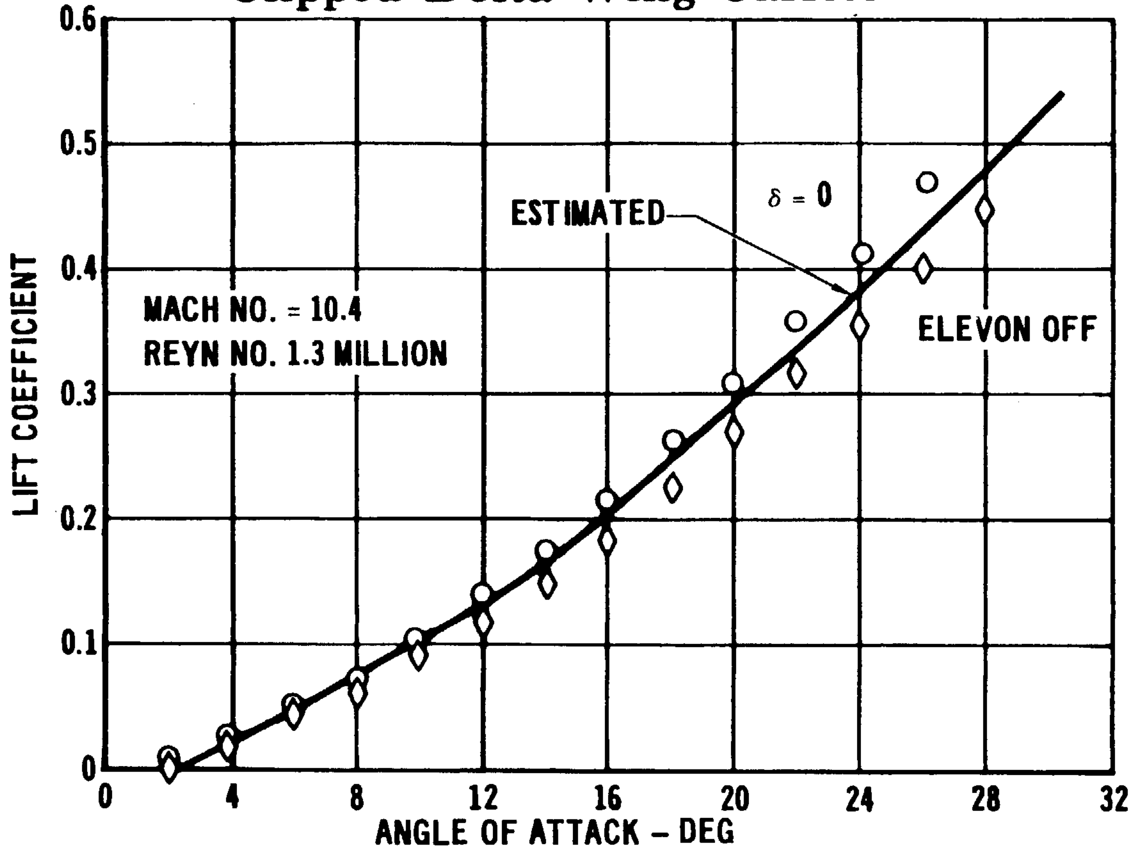


ILRVS-461F

HYPERSONIC LIFT CURVE

The hypersonic test data which shows the lift characteristics of Clipped Delta Wing Carrier is presented in the figure on the adjacent page. Two configurations were tested; i.e., the vehicle with the elevons at zero deflection angle, and the vehicle with elevon control surfaces removed. A hypersonic estimate for the vehicle with zero control deflection has been superimposed on the test data for comparison pruposes. The estimate was generated with the Hypersonic Arbitrary Body program, utilizing standard Newtonian theory on the windward surfaces and Prandtl-Meyer expansion techniques in the leeward surfaces. The vehicle reference area is the theoretical wing area.

LIFT CURVE
Clipped Delta Wing Carrier

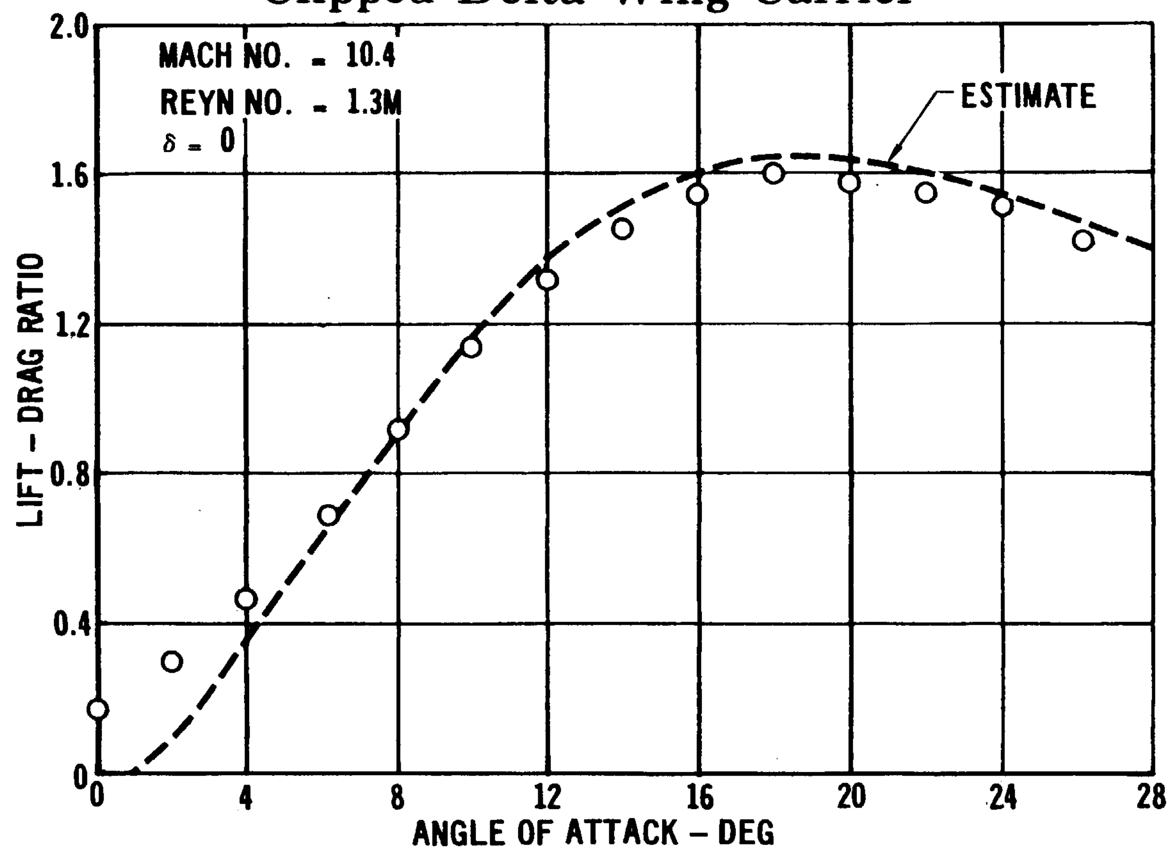


ILRVS-456F

HYPERSONIC LIFT-DRAG RATIO

The next figure presents the tested hypersonic L/D characteristics for the Clipped Delta Wing Carrier. The data is for the configuration with zero control deflection and indicates a maximum value of 1.6 at 18° angle of attack. An estimated L/D, generated with the Hypersonic Arbitrary program, has been added to the figure and shows favorable agreement at all but the lowest angles of attack. The L/D data presented here was taken directly from the test results and presently includes no corrections for scale effects.

LIFT DRAG RATIO
Clipped Delta Wing Carrier

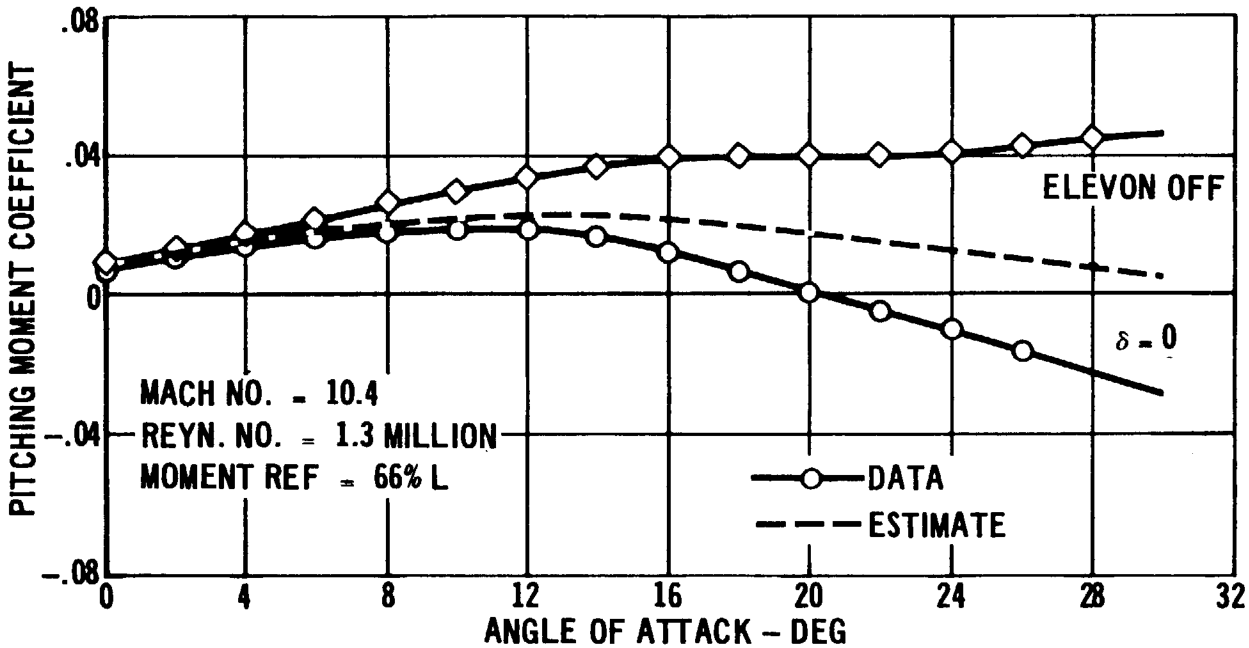


ILRVS–458 F

HYPERSONIC PITCHING MOMENT

The figure on the following page presents the tested hypersonic pitching moment characteristics for the Clipped Delta Wing Carrier. The moment reference point is at the 66% station on the vehicle centerline. The data is shown for two carrier configurations; i.e., the vehicle with zero control deflection, and the vehicle with control surface removed. An estimated hypersonic pitching moment coefficient for the vehicle with zero control deflection was included in the figure for comparison purposes. The estimate was obtained through the use of the Hypersonic Arbitrary Body program utilizing standard Newtonian theory in conjunction with Prandtl-Meyer expansion techniques. The reference area is the theoretical wing area and moment reference length is the corresponding mean aerodynamic chord.

PITCHING MOMENT
Clipped Delta Wing Carrier

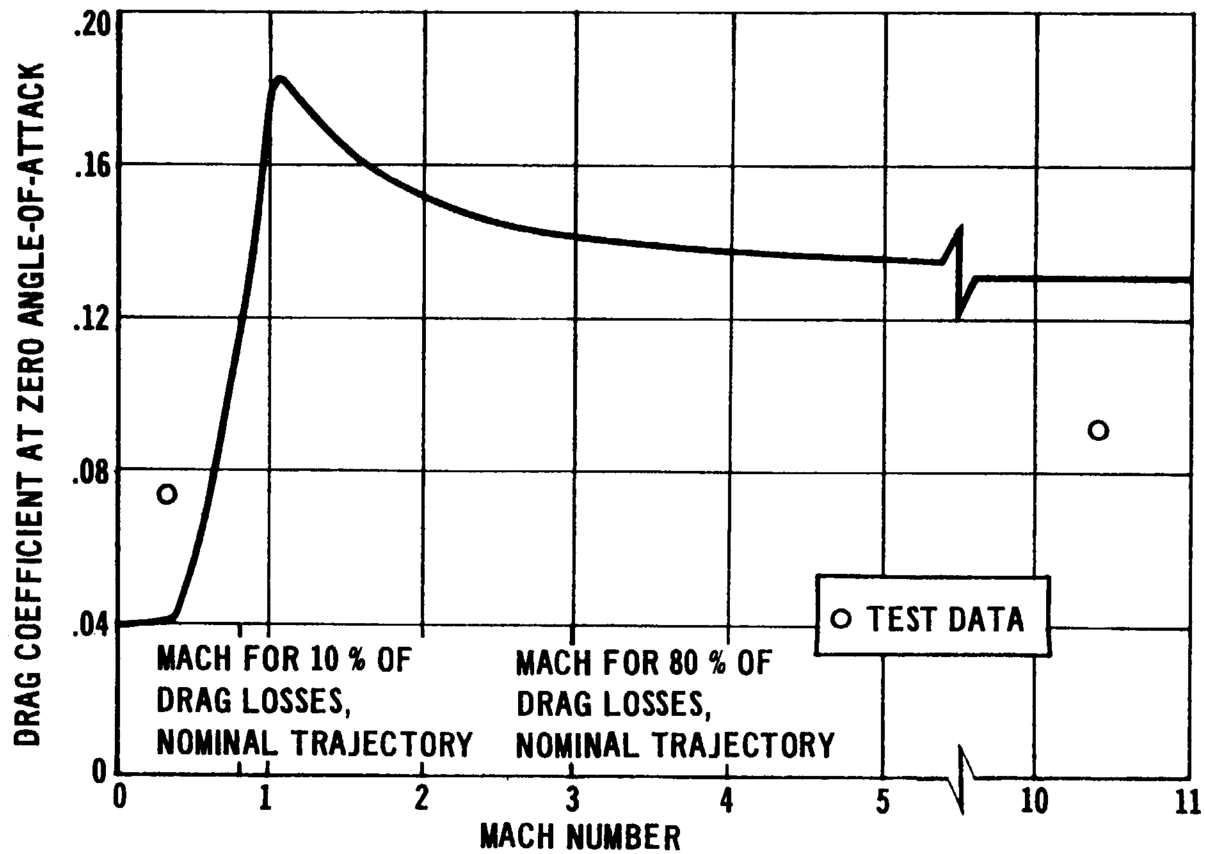


ILRVS–459F

ASCENT CONFIGURATION DRAG COEFFICIENT

The following figure shows the estimated zero angle of attack drag coefficient variation, for the Ascent Configuration, across the Mach number range. The ascent configuration consists of the first stage delta wing carrier attached to the HL-10 orbiter. Two test data points have been included in the figure and indicate that the estimates were low in the subsonic range and high in the hypersonic range. However, the test data does not include base pressure corrections due to engine thrust effects and may not be a true indicator of the actual boost phase drag. The figure also shows the Mach number range where the majority of the drag losses are accumulated and thus delineates the area which must be well defined for accurate drag loss predictions. The reference area for the ascent configuration is the theoretical wing area of the clipped delta wing carrier.

ASCENT CONFIGURATION DRAG COEFFICIENT

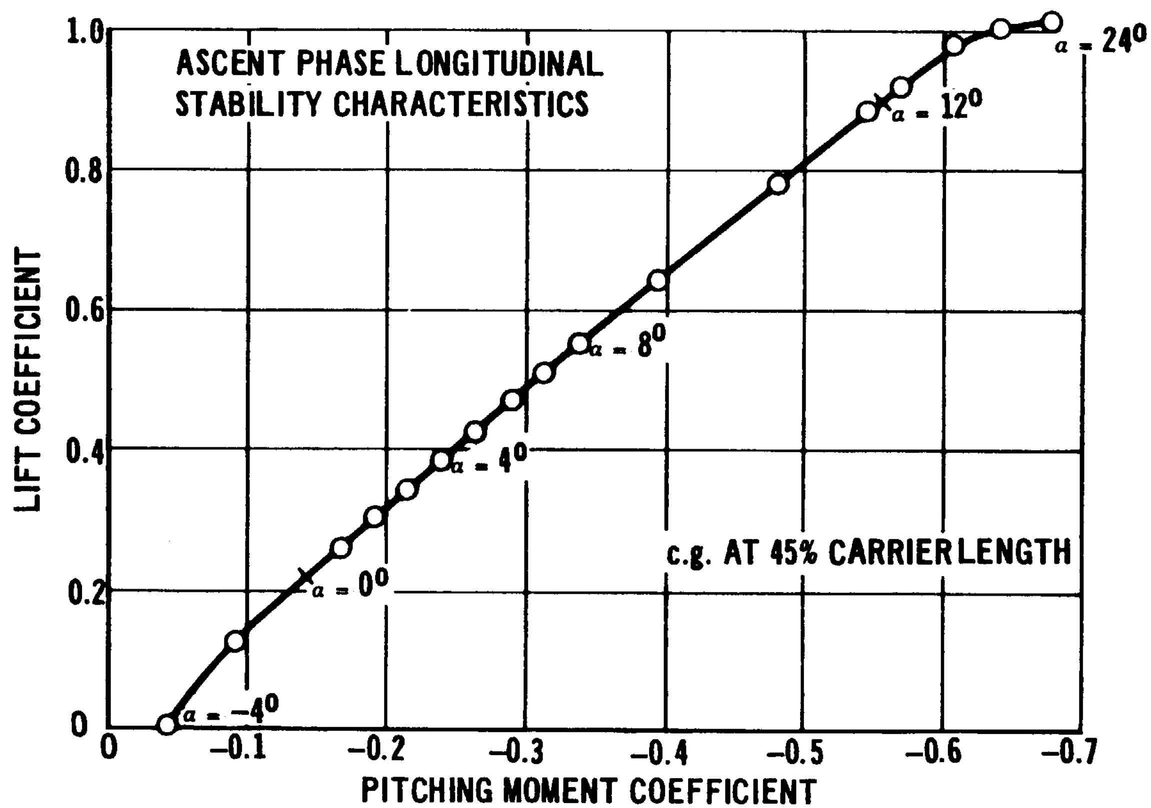


ILRVS-468 F

ASCENT PHASE LONGITUDINAL STABILITY CHARACTERISTICS

The adjacent figure shows the tested static stability characteristics for the ascent configuration. The ascent configuration consists of the first stage delta wing carrier attached to the HL-10 orbiter. The moment reference point is at the 45% station, which represents the combined center of gravity of the fully loaded carrier and HL-10. The figure indicates that the configuration is stable and the addition of elevon control deflections would also show that it is possible to trim the vehicle. However, this is true for the power-off condition only, since the power-on trim condition would be drastically affected by the thrust moments of the engines. The reference area is the carrier theoretical wing area and the moment reference length is the corresponding mean aerodynamic chord.

SUBSONIC TEST DATA



ILRVS-475F

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REENTRY HEATING

AND

THERMAL PROTECTION

Special Emphasis Area

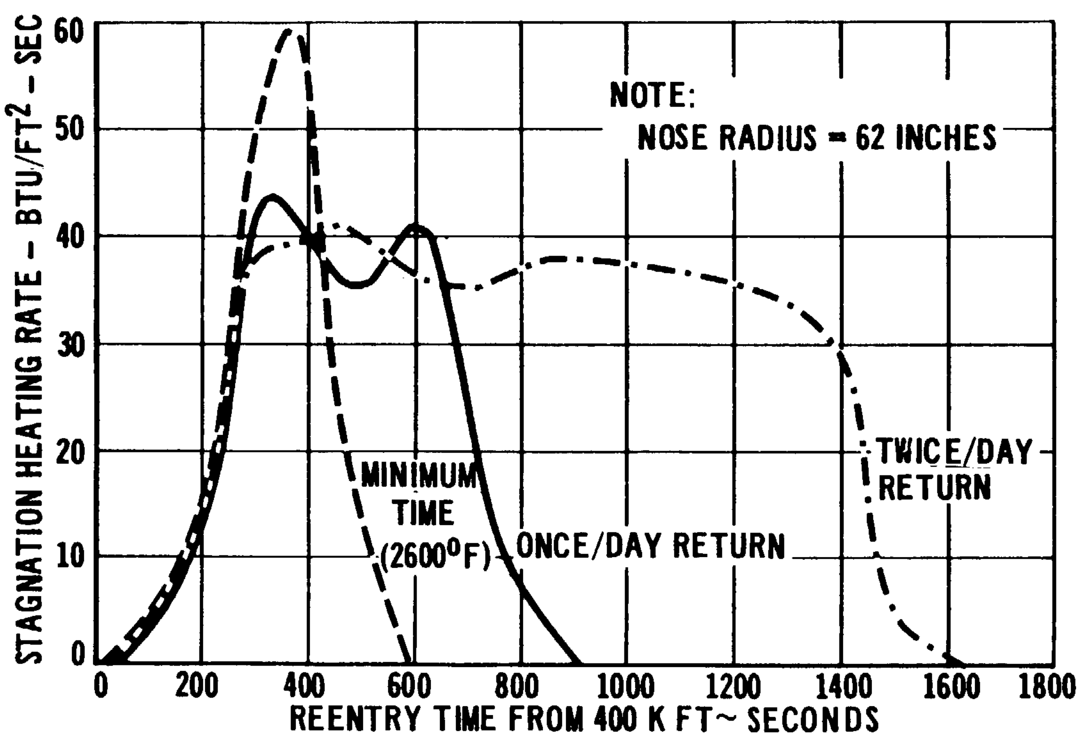
ILRVS-436F

REENTRY HEAT PROFILES - ORBITER

Heat pulses for HL-10 reentries are shown for a stagnation point having a 62 inch radius. Heat pulses are shown for the once/day, Minimum Time (2600°F) and twice/day reentries. The twice/day reentry incurrs the largest amount of total heat but the lowest heating rate while the minimum time (2600°F) reentry incurrs the lowest amount of total heat but the highest heating rate.

The heating rates are based on the Fay and Riddell stagnation point theory and a constant wall temperature of 2000°R.

REENTRY HEAT PROFILES ORBITER



ILRVS-350F

ORBITER MAXIMUM TEMPERATURES
ONCE/DAY TRAJECTORY

Maximum surface temperatures experienced during once/day reentry are shown for four HL-10 body stations.

Heating rates were determined using the stagnation point heat pulse previously shown and NASA-LRC heat transfer data.

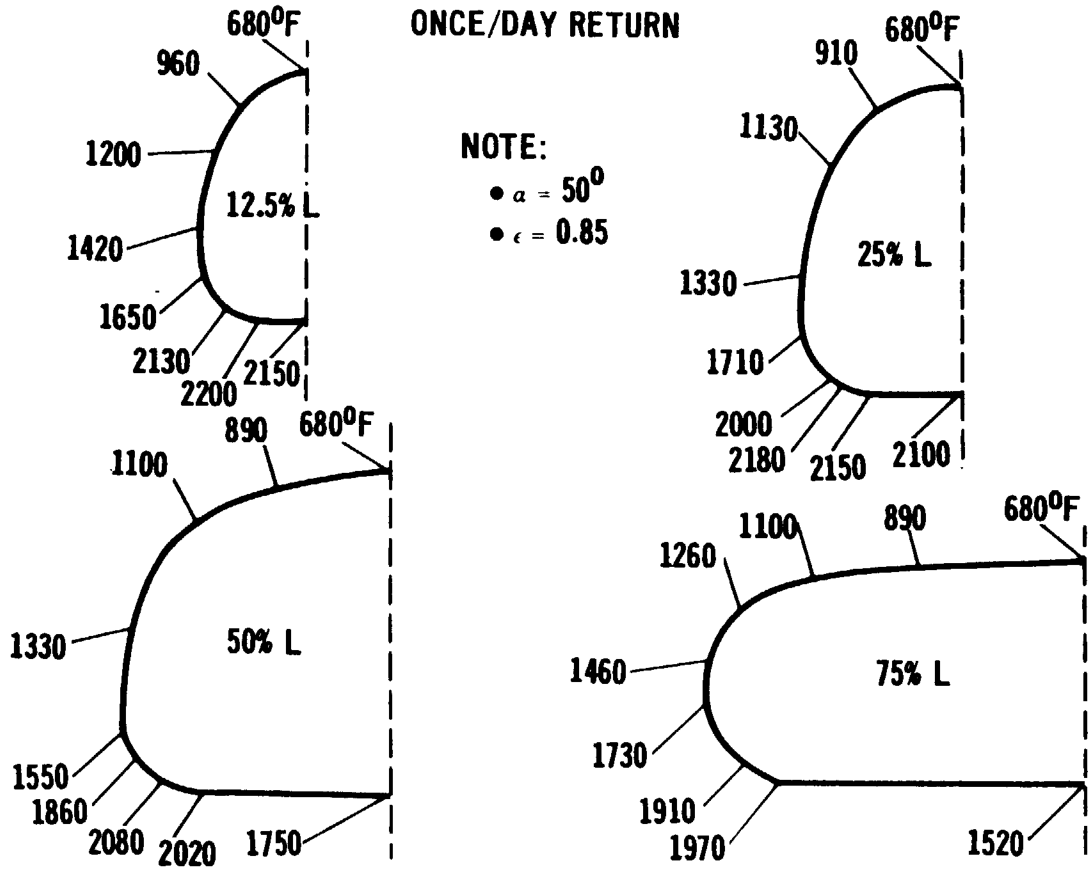
Temperatures shown are radiation equilibrium values based on a surface emittance equal to 0.85 and occur at the time of the peak heating rate of 44 BTU/ft^{2}sec. previously shown.

Surface temperatures range from 680°F on the vehicle upper surface to 2200°F on the lower surface.

HL–10 TEMPERATURES

ONCE/DAY RETURN

NOTE:
- $\alpha = 50^0$
- $\epsilon = 0.85$



ILRVS–351

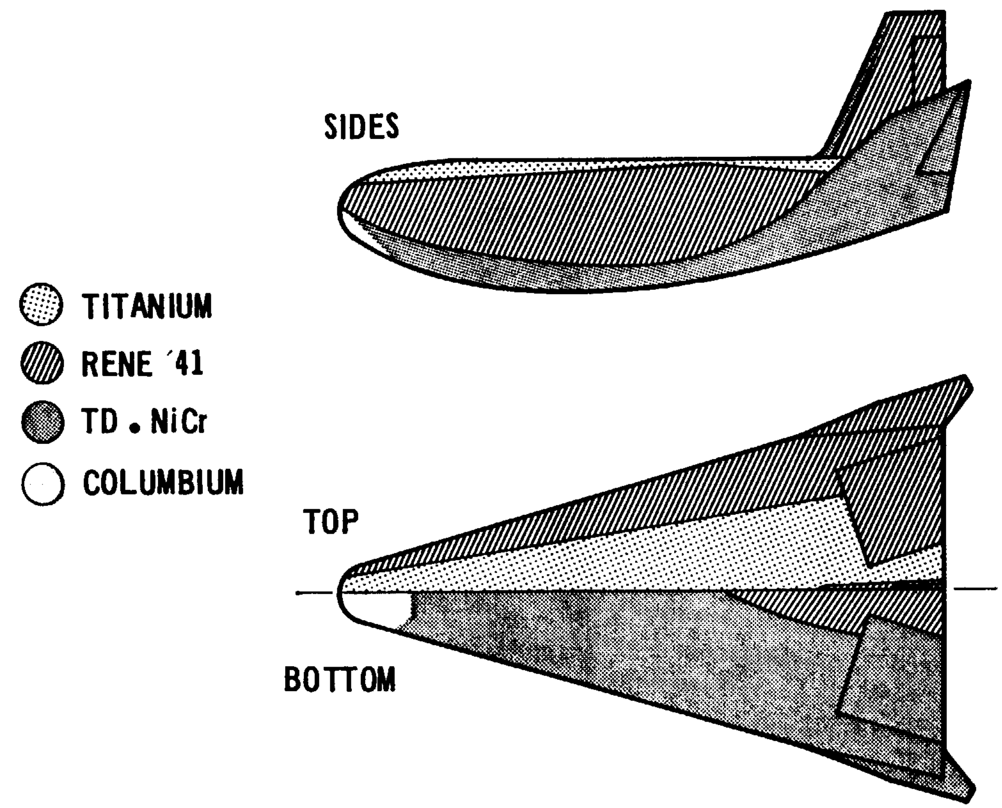
MATERIAL DISTRIBUTION OF TPS SHINGLES
HL-10
(Metallic Shingles)

Material selection for TPS shingles is based on the following temperature use ranges:

Titanium	400-1000°F
René 41	1000-1600°F
TD-NiCr	1600-2200°F
Columbium	2200-2900°F

The material distribution shown is derived from the temperature distribution shown earlier.

MATERIAL DISTRIBUTION OF TPS SHINGLES HL-10



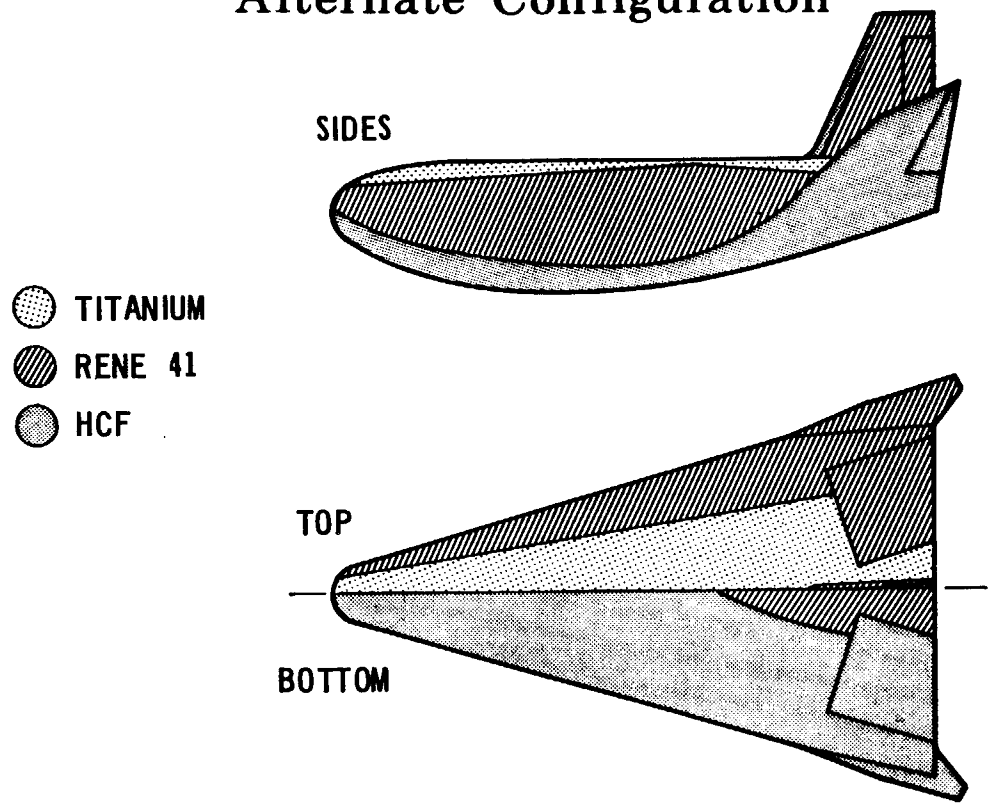
ILRVS–317F

MATERIAL DISTRIBUTION OF TPS SHINGLES
HL-10
(Metallic and Non-Metallic Shingles)

This arrangement is an alternative to the metallic shingle, in regions between 1600° and 2600°F, with HCF supported by a fiberglass honeycomb sub-structure. Temperature use ranges are:

Titanium	400-1000°F
René 41	1000-1600°F
HCF	1600-2600°F

MATERIAL DISTRIBUTION OF TPS SHINGLES HL-10
Alternate Configuration



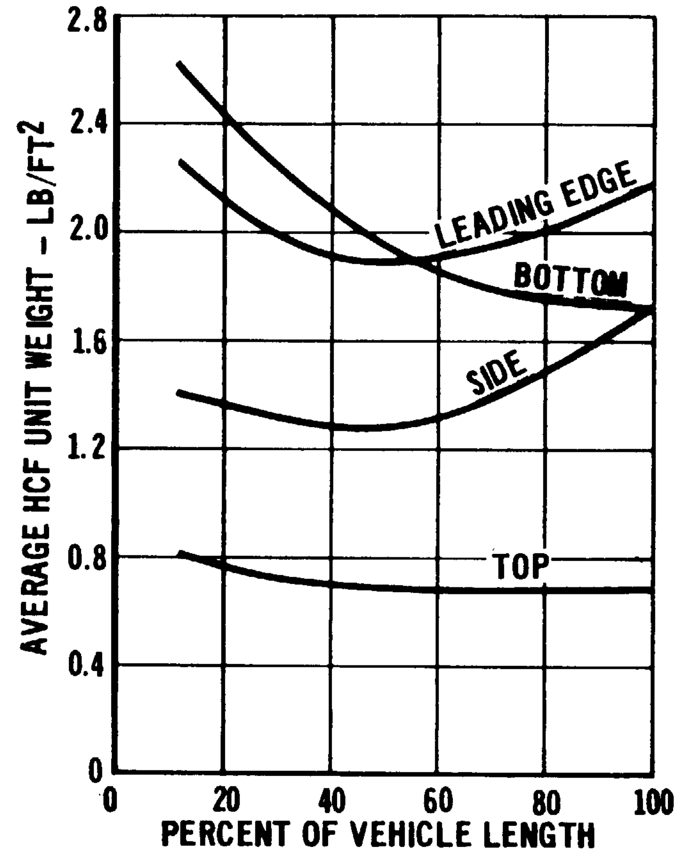
ILRVS-318F

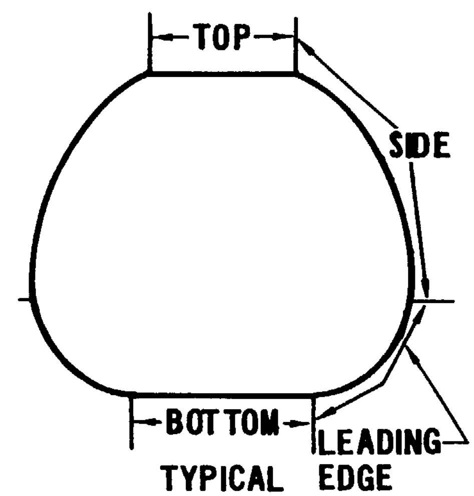
HCF UNIT WEIGHTS - ORBITER
MINIMUM TIME (2600°F) TRAJECTORY

HCF unit weight requirements are shown for the minimum time (2600°F) HL-10 reentry. These requirements are based on a maximum bondline temperatures of 500°F and a material density of 15 lb/ft^3.

Unit weight requirements vary from 2.6 lb/ft^2 on the lower surface to 0.7 lb/ft^2 on the upper surface. Because of the relatively high heating rates on the sides of the HL-10 large applications of HCF are required.

HCF UNIT WEIGHTS-ORBITER
Minimum Time (2600°) Trajectory





ILRVS-354F

NOTE:
- P_{HCF} = 15 LB/FT^3
- 500^{0}F MAX BONDLINE

THERMAL PROTECTION SYSTEM WEIGHT COMPARISON

This chart illustrates a point analysis of the thermal protection system weight. The point chosen for this example is X/L = .25 or 26.7 ft. from the nose on the bottom centerline. The baseline design and reference point is TD-NiCr shingles. The unit weight data shown on the chart indicates that the baseline system is not the lightest available. However, the total thermal protection system weight is the sum of numerous analyzed points which will reflect the change in temperatures and subsequent unit weight changes along the vehicle surface. This net effect is shown in the sums (Δ CARGO and Δ GLOW). From these it is seen that the baseline system is the lightest.

THERMAL PROTECTION SYSTEM
UNIT WEIGHT COMPARISON

- HL-10
- X/L = 0.25 (LOWER ℄)

· ENTRY TRAJECTORY	TWICE/DAY RETURN		ONCE/DAY RETURN		MIN TIME (2600°F)
SYSTEM COMPONENT	TDNi Cr (LB)	HCF (LB)	TDNi Cr (LB)	HCF (LB)	HCF (LB)
TPS	(2.75)	(4.12)	(3.30)	(3.55)	(3.45)
●SHINGLE	1.10	2.95	2.10	2.41	2.29
●FIBERGLASS PANEL		0.54		0.51	0.53
●STRAP	0.10		0.10		
●ATTACH	0.42	0.24	0.42	0.24	0.24
●INSULATION	1.03	0.29	0.58	0.29	0.29
●INSUL. ATTACH	0.10	0.10	0.10	0.10	0.10
STRUCTURE (FRAME)	(2.20)	(1.40)	(1.90)	(1.40)	(1.40)
TOTAL UNIT WT.	4.95	5.52	5.20	4.75	4.80
Δ CARGO	−3,700	−17,500	REF.	−1,500	−1,000
Δ GLOW	+ 149,900	+ 746,000	REF.	+ 60,600	+ 41,270

TPS STATE-OF-ART COMPARISON

Much more is known about the material properties of TD-NiCr (being studied by MDAC under Air Force contract) than HCF. Since mechanical properties are of secondary importance for HCF, it does not really lag behind TD-NiCr.

Although the values shown indicate a difficiency in being able to come up with applicable attachment and joining techniques, here again there is less actually needed for effective design utilization.

Scale up has been demonstrated for both but it is believed that fabrication of HCF materials will be more complicated as borne out by the higher complexity factors (based on Al = 0.2).

Development funding for both will have to be increased if either material is to find application on shuttle type vehicles. Development of TD-NiCr would more closely follow conventional metal development whereas HCF being a material having more exotic characteristics would require development of more unique manufacturing techniques.

Of the two materials, TD-NiCr has the better chance of successfully meeting shuttle objectivves. However, there is higher confidence that coated Columbium can not only be employed but can be done with less development, both in time and money. MDAC has extensive experience with this material having successfully flown it on both ASSET and BGRV. Reuse testing on fused slurry coated Cb752 is presently under Air Force contract funded study. Preference at this time is for utilization of coated Columbium as the primary lower surface best shield material.

TPS STATE-OF-ART COMPARISON

AREA OF INTEREST	STATUS		
	HCF	TD NiCr	COLUMBIUM
MATERIAL PROPERTIES			
MECHANICAL	≈ 30%	≈ 75%	≈ 95%
THERMAL	≈ 30%	≈ 75%	≈ 95%
ATTACHMENT JOINING TECHNIQUES	≈ 30%	≈ 75%	≈ 95%
SCALE-UP	DEMCNSTRATED	DEMONSTRATED	DEMONSTRATED
LARGE SCALE TESTING	SATURN BASE HEATING*	IN WORK**	FIN-RUDDER, ASSET, BGRV
FLIGHT RE-USE TESTING	NONE	NONE	ASSET-BGRV
PRESENT DEV. FUNDING	VERY LOW	LOW	MODERATE
GRD HANDLING RESISTANCE	REQUIRES PROTECTION	ADEQUATE	ADEQUATE
NORMAL MFG TECHNIQUES	NO	YES	YES (DEMONSTRATED)

* HONEYCOMB FILLED WITH AN HCF TYPE MATERIAL
** VERTICAL FIN STRUCTURAL ASSEMBLY

ILRVS-375F

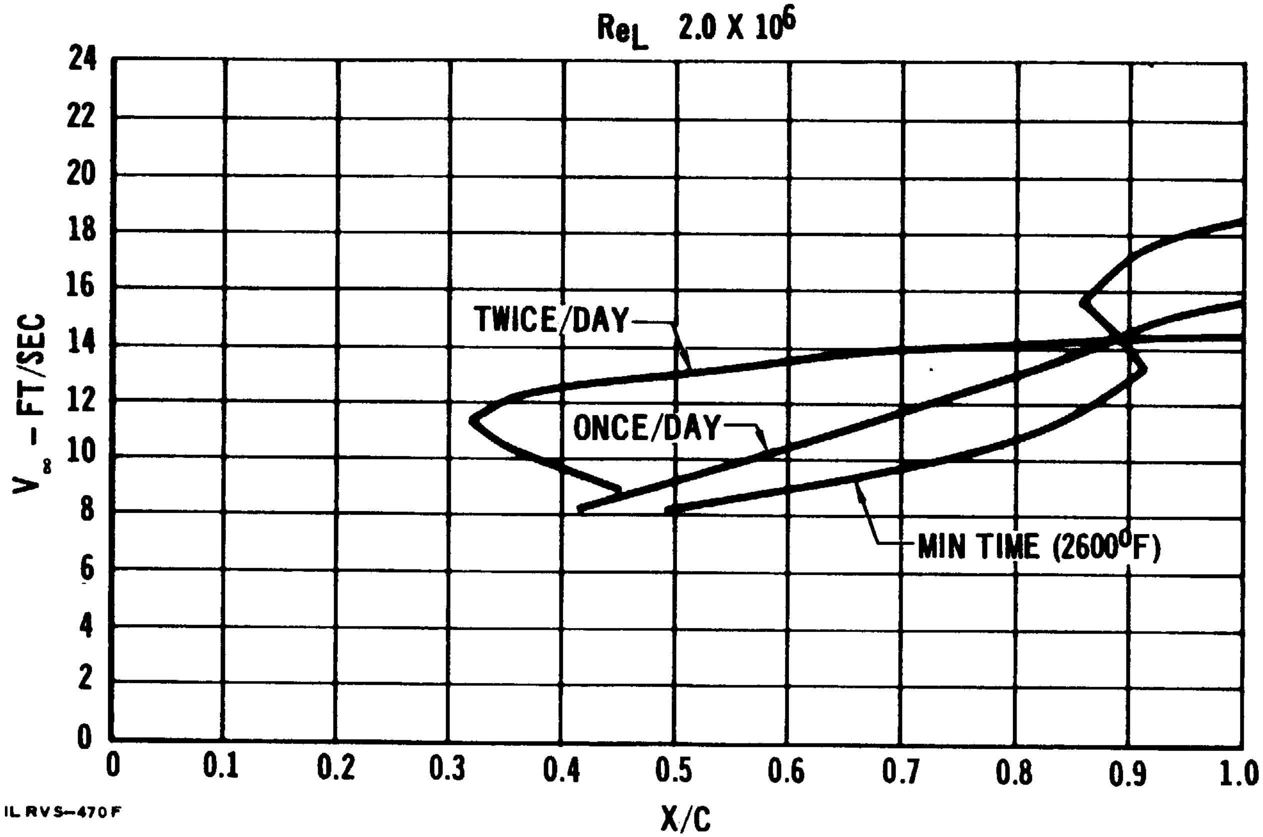
TRANSITION EFFECTS ON ORBITER TEMPERATURES
FULLY DEVELOPED TURBULENT FLOW

Velocities at which fully developed turbulent flow occurs along the HL-10 lower surface centerline are shown for the once/day, twice/day and minimum time 2600°F reentries. These velocities occur at $Re_L = 2.0 \times 10^6$.

Except for the aft 10 percent vehicle length transition occurs at higher velocities for the twice day reentry than for either minimum time reentry.

Complete transition to turbulent flow occurs at 550, 770 and 1410 seconds from initiation of reentry for the minimum time 2600°F, once/day and twice/day reentries respectively.

FULLY DEVELOPED TURBULENT FLOW



ILRVS-470F

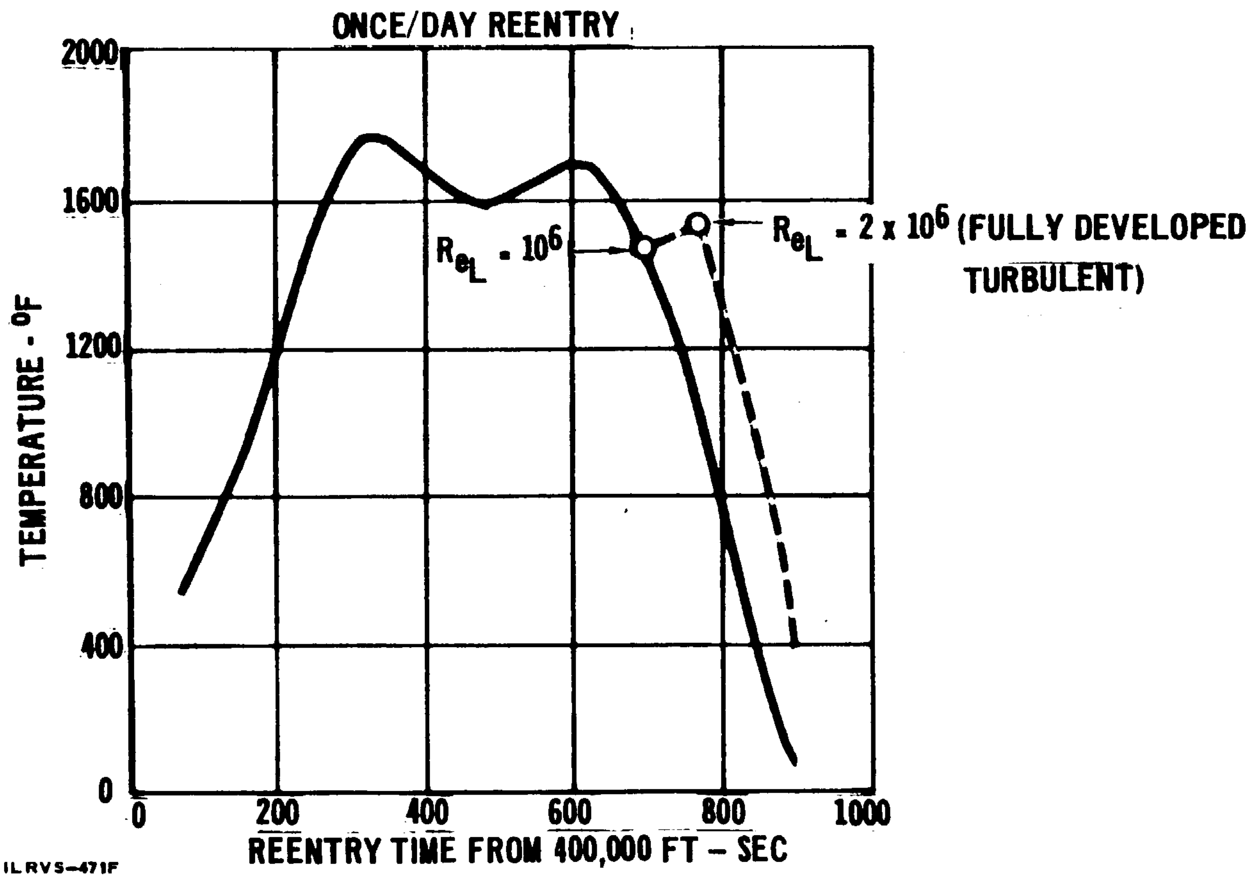
TRANSITION EFFECTS ON ORBITER TEMPERATURES

The effect of boundary layer transition to turbulent flow on HL-10 lower surface centerline temperatures at the 50% body length are shown for the once/day reentry.

Transition onset occurs at $Re_L = 1.0 \times 10^6$ and fully developed turbulent flow occurs at $Re_L = 2.0 \times 10^6$.

The maximum turbulent temperature (1500°F) is lower than the maximum laminar temperature (1750°F) because fully developed turbulent flow occurs late flight at a low velocity (9400 ft/sec).

TRANSITION EFFECTS ON ORBITER TEMPERATURES



ILRVS–471F

EXPERIMENTAL HEATING

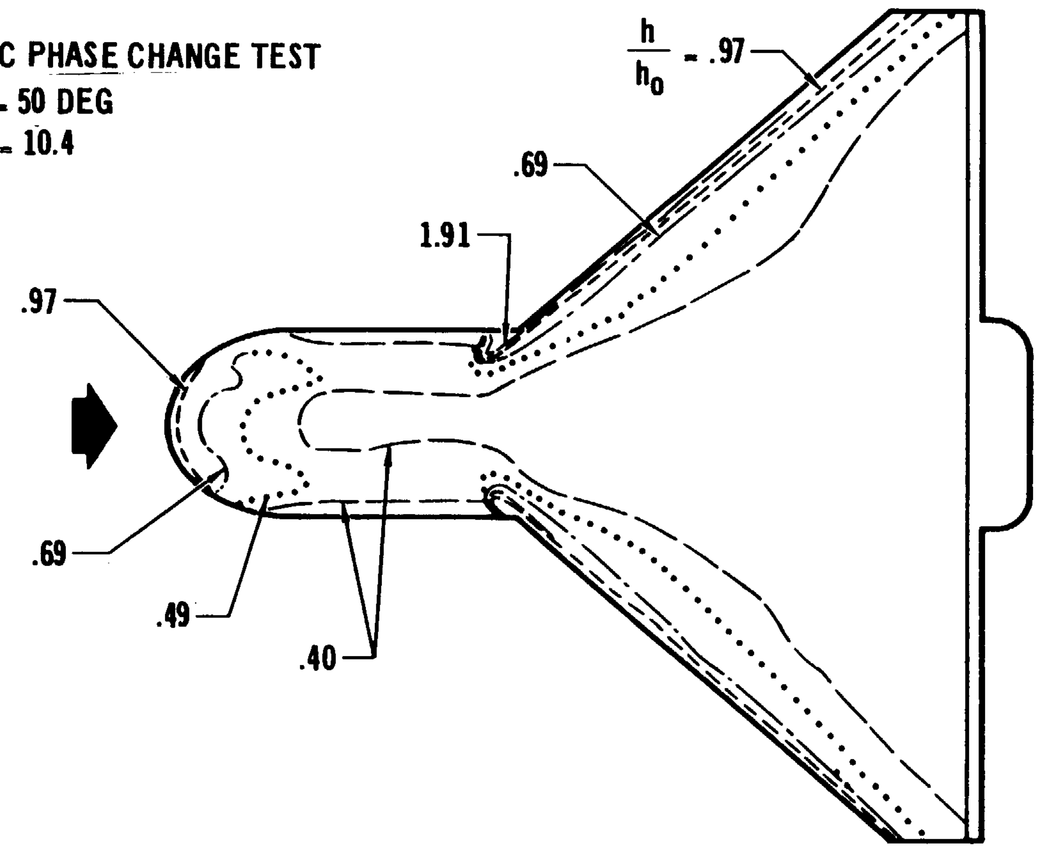
Experimental heat transfer tests were conducted by NASA-LRC on a low wing clipped delta configuration. Tests were conducted at a Mach number of 10.4 and a Reynolds number of 0.5×10^6 based on model length. The model was coated with a phase change material and local heating rates were determined by interpretation of photographic data.

The figure shows lines of constant heating rates as interpreted from the photographic data. The values shown are ratios of local heating rates to a calculated stagnation point heating rate on a hemisphere having a diameter equal to the vertical thickness of the model.

Tests results indicate that the high heating rates at the wing root-body juncture is reduced by the fairing incorporated in the baseline shape.

EXPERIMENTAL HEATING

LRC PHASE CHANGE TEST
α = 50 DEG
M = 10.4



ILRVS-462F

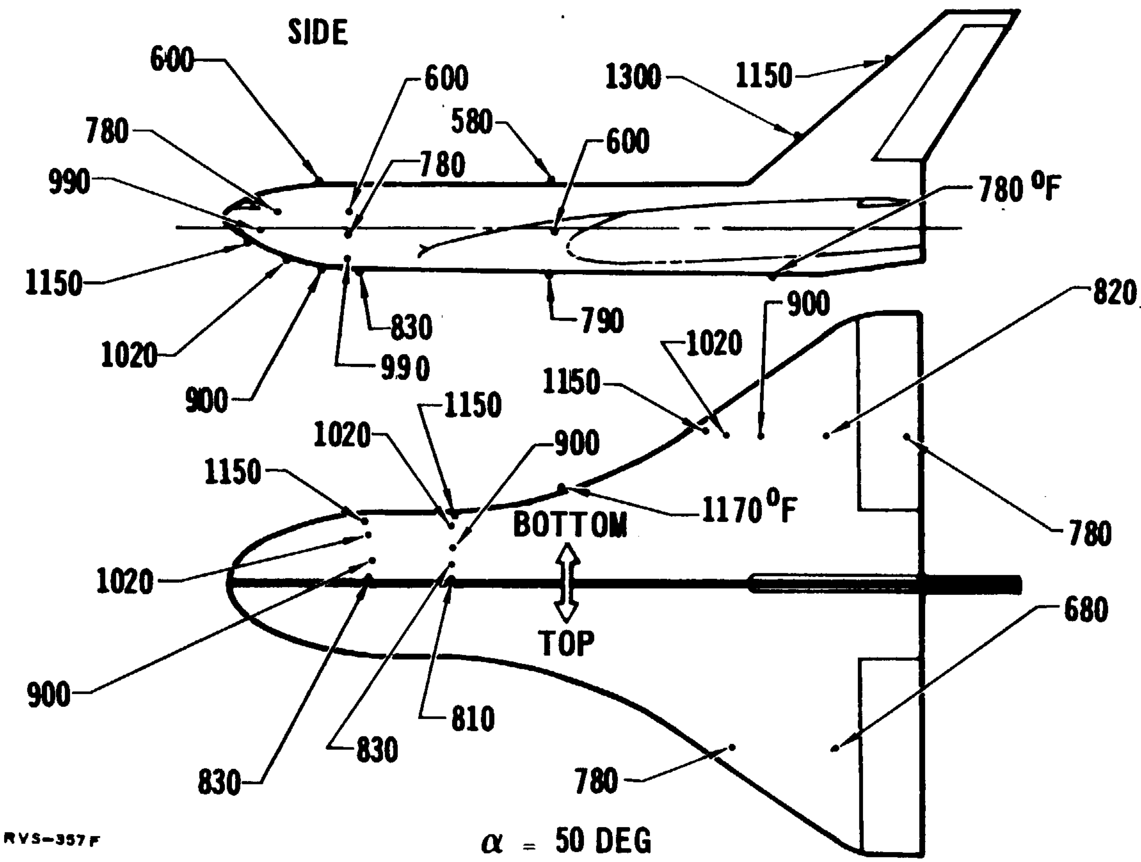
MAXIMUM CARRIER TEMPERATURES

Maximum temperatures experienced by the carrier during launch and reentry are shown. Upper surface and fin maximum temperatures are reached during launch at low angle of attack.

The temperatures correspond to an ideal ΔV of 15,000 ft/sec and were estimated utilizing the NASA-LRC phase change test data.

Maximum lower surface temperatures approach 1200°F and the upper surface temperatures vary from 600°F to 800°F.

MAXIMUM CARRIER TEMPERATURES



ILRVS-357F

MATERIAL DISTRIBUTION OF TPS SHINGLES

CARRIER

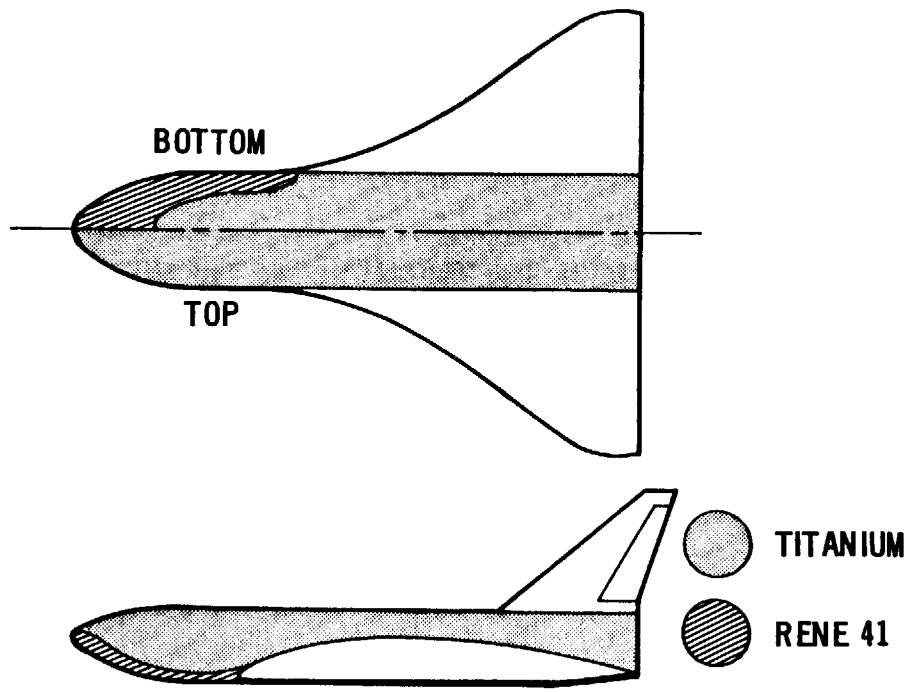
Material selection is based on the following temperature use ranges:

Titanium	400-1000°F
René 41	1000-1600°F

Non-structural shingles are used only on the body external surface. Wing and vertical tail surface panels are structural and are also fabricated from Titanium and René 41 alloy materials.

MATERIAL DISTRIBUTION OF TPS SHINGLES CARRIER

NOTE: WING & VERTICAL TAIL SURFACE PANELS
ARE STRUCTURAL (TITANIUM WITH RENE LEADING EDGES)



ILRVS-316F

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ABORT

Special Emphasis Area

ILRVS-438F

ABORT PHILOSOPHY

Commercial airline operational philosophy concerning abort is carried over to the ILRVS in the spacecraft's integrity. This integrity is assured by designing in multiple redundancy in all major subsystems affecting the safety of the vehicle and hence, the crew and passengers. Examples are: engine out capability in both stages; all mechanical systems designed to fail operationally and then fail safe. Avionics systems design will allow two fail operational modes plus fail safe.

"Holddown" capability on the pad prior to liftoff will be provided along with a quick egress system. On board crew control of possible abort situations through display panels will allow rapid assessment and reaction to non-normal situations.

Intact abort capability throughout all missions phases except for a small deadband just after liftoff will be possible.

ABORT PHILOSOPHY

SPACECRAFT AND PERSONNEL INTEGRITY ASSURED BY –

- MULTIPLE REDUNDANCY
  PROPULSION – ENGINE OUT CAPABILITY, BOTH STAGES
  MECHANICAL – FAIL OPERATIONAL/FAIL SAFE
  AVIONICS – FAIL OPERATIONAL (TWICE)/FAIL SAFE
- "HOLD DOWN" CAPABILITY PRIOR TO LIFTOFF
- ON PAD QUICK EGRESS SYSTEM
- INTACT ABORT CAPABILITY DURING ALL MISSION PHASES
- ON-BOARD CREW CONTROL OF POSSIBLE ABORT SITUATIONS

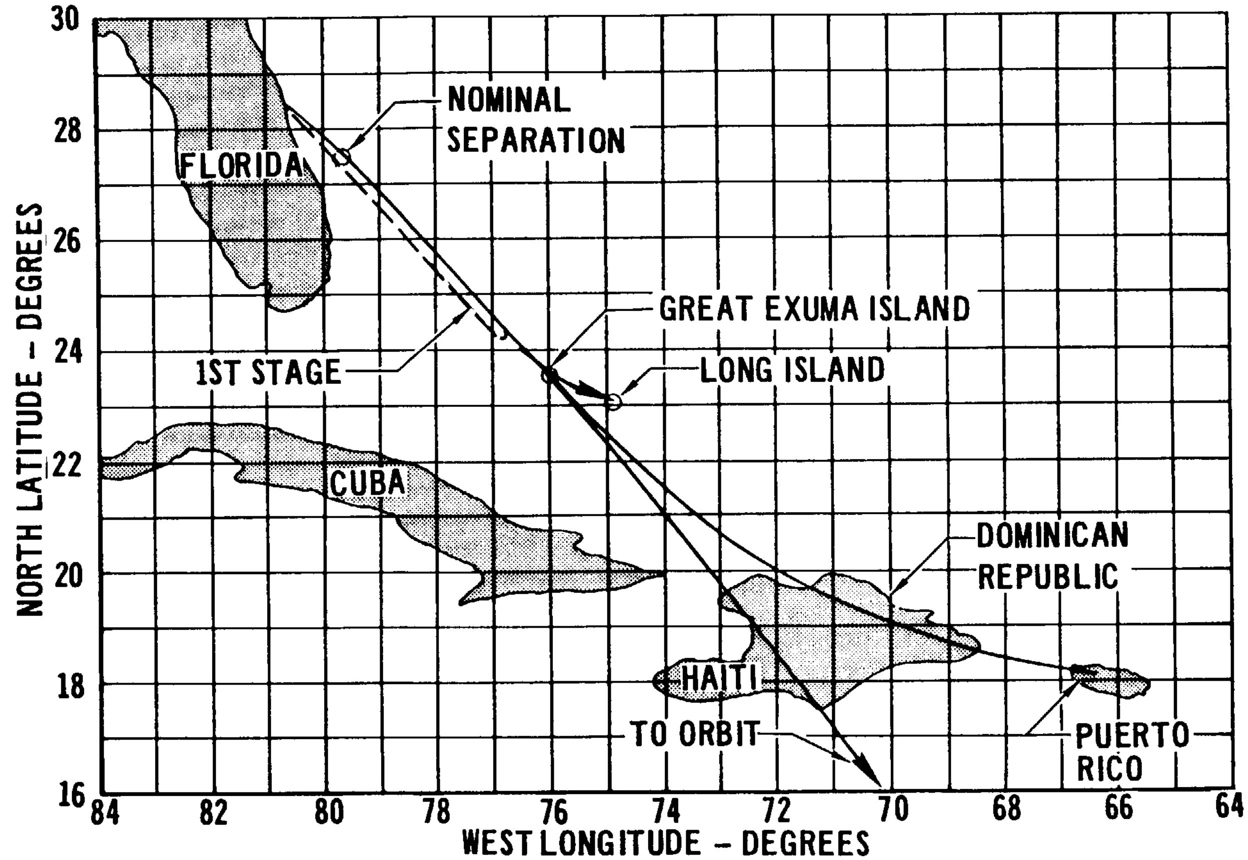
ILRVS-489F

ABORT LANDING SITES

Potential landing sites in case of abort during a southerly launched mission are shown on the opposite page. As shown, the nominal 55° inclination ground track passes almost directly over Great Exuma Island and Long Island. These sites were because of their downrange location as well as cross range location. In the event of failure of the orbit engines to ignite at nominal separation, the HL-10 orbiter can reach either of these islands. If abort occurs during carrier action, the performance of the HL-10 orbiter is sufficient to achieve these same separation conditions and recover at the alternate landing sites.

Landing at Puerto Rico is indicated for aborts which occur for velocities in the range of 14700 to 17400 feet per second. Since the carrier velocity capability is 9166 feet per second it is required that the orbiter engines provide the additional velocity to achieve these velocities. The upper and lower velocity limit is established by the energy management capability of the orbiter; i.e. L/D max and L/D min. These burnout conditions can also be achieved by the orbiter for abort from the carrier after a velocity of 4200 ft/sec has been achieved. Residual fuel weight at burnout increases as abort occurs closer to nominal carrier burnout.

ABORT LANDING SITES

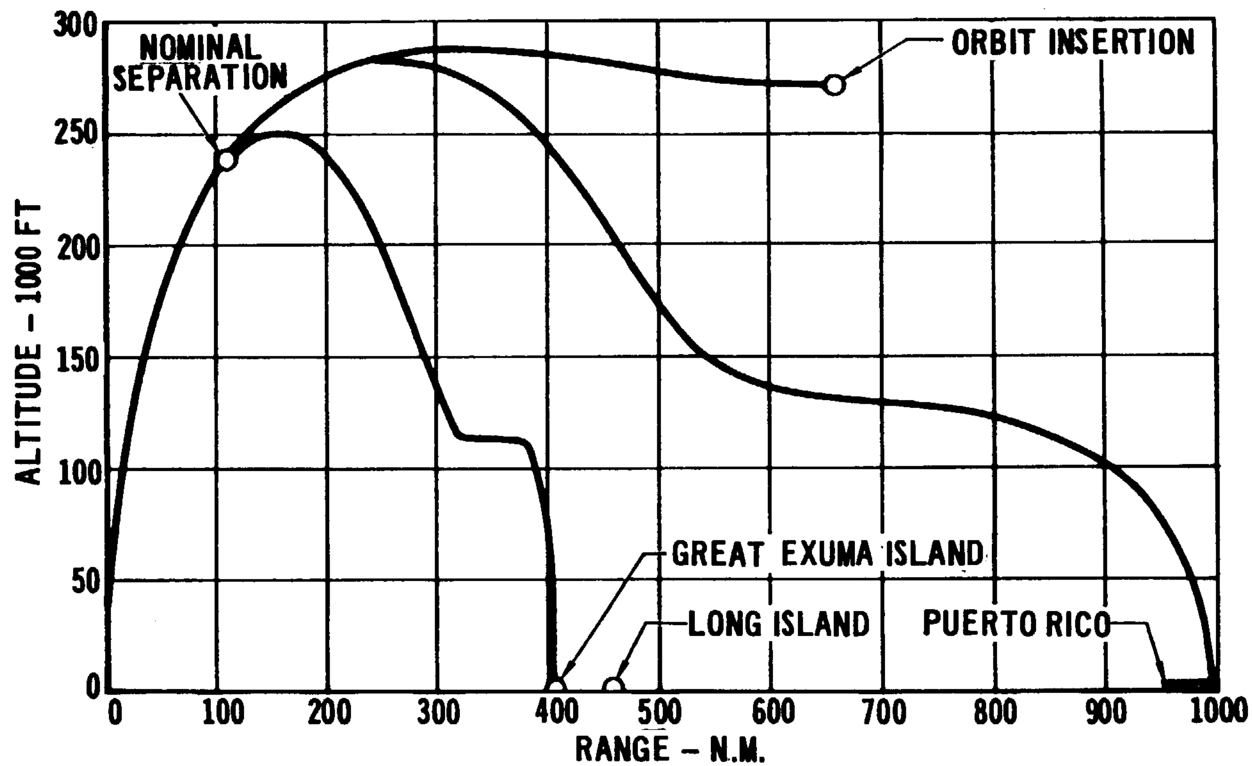


ILRVS – 415F

ABORT TRAJECTORY PROFILES

The opposing altitude range profiles show nominal trajectories and three possible abort trajectories.

ABORT TRAJECTORY PROFILES



ILRVS-417F

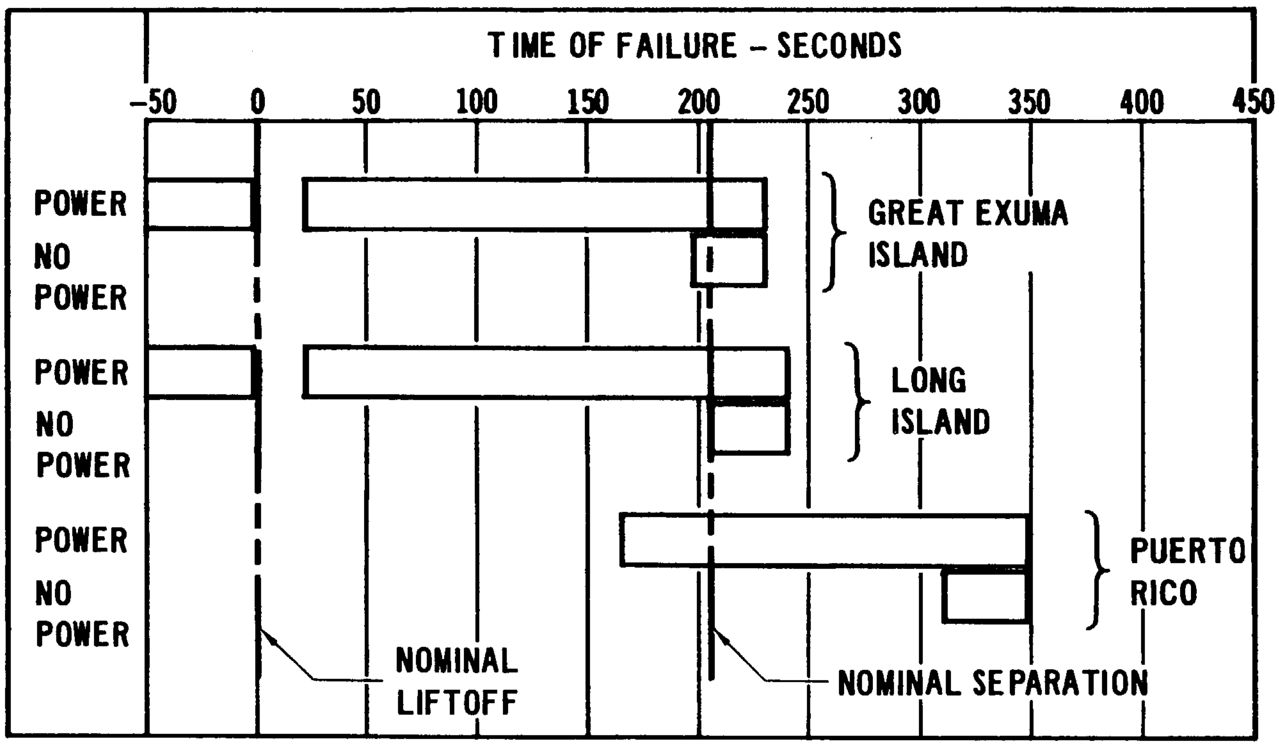
ORBITER ALTERNATE ABORT MODES

The primary abort mode is abort to orbit. Provisions have been made for one-engine-out conditions for carrier as well as orbiter. However, in the event of a double failure mode intact recovery of the orbiter can be accomplished.

The opposing figure indicates which landing sites can be chosen for failure at a particular time. For the first 20 seconds of flight intact recovery of the orbiter is improbable. After 350 seconds the orbiter can recover and do a safe pullout but a landing site is not available. For failure at any other time, a safe landing at an airport is possible if orbiter power is available.

Without orbiter power, airport recovery is possible if failure occurs in the first 30 seconds after separation or between times of 310 and 350 seconds. Failure at other times will require a water landing and air sea rescue.

ORBITER ALTERNATIVE ABORT MODES



IL RVS-416F

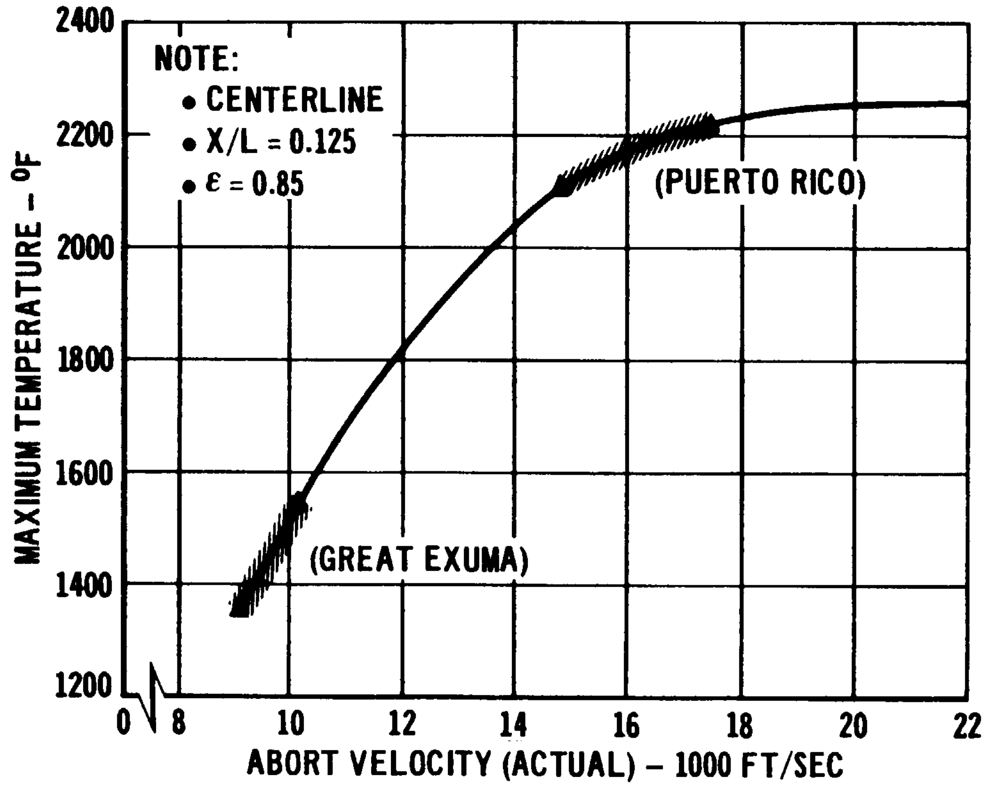
ORBITER ABORT LOWER SURFACE TEMPERATURE

Maximum temperatures on the lower surface at 12.5% vehicle length are shown for abort. The maximum temperatures range from 1300°F for an abort velocity of 8800 ft/sec to 2260°F for an abort velocity of 22000 ft/sec.

Abort to a landing at Grand Exuma occurs at a velocity of 9166 ft/sec and results in a maximum temperature of approximately 1350°F. Abort to a landing at Puerto Rico occurs at a velocity of 13700 ft/sec and results in a maximum temperature of approximately 2020°F.

Abort can be accomplished to Great Exuma, Puerto Rico or orbit injection from any point after T_o + 20 sec. or off the pad without exceeding the design temperature limits of the orbiter.

ORBITER ABORT TEMPERATURES

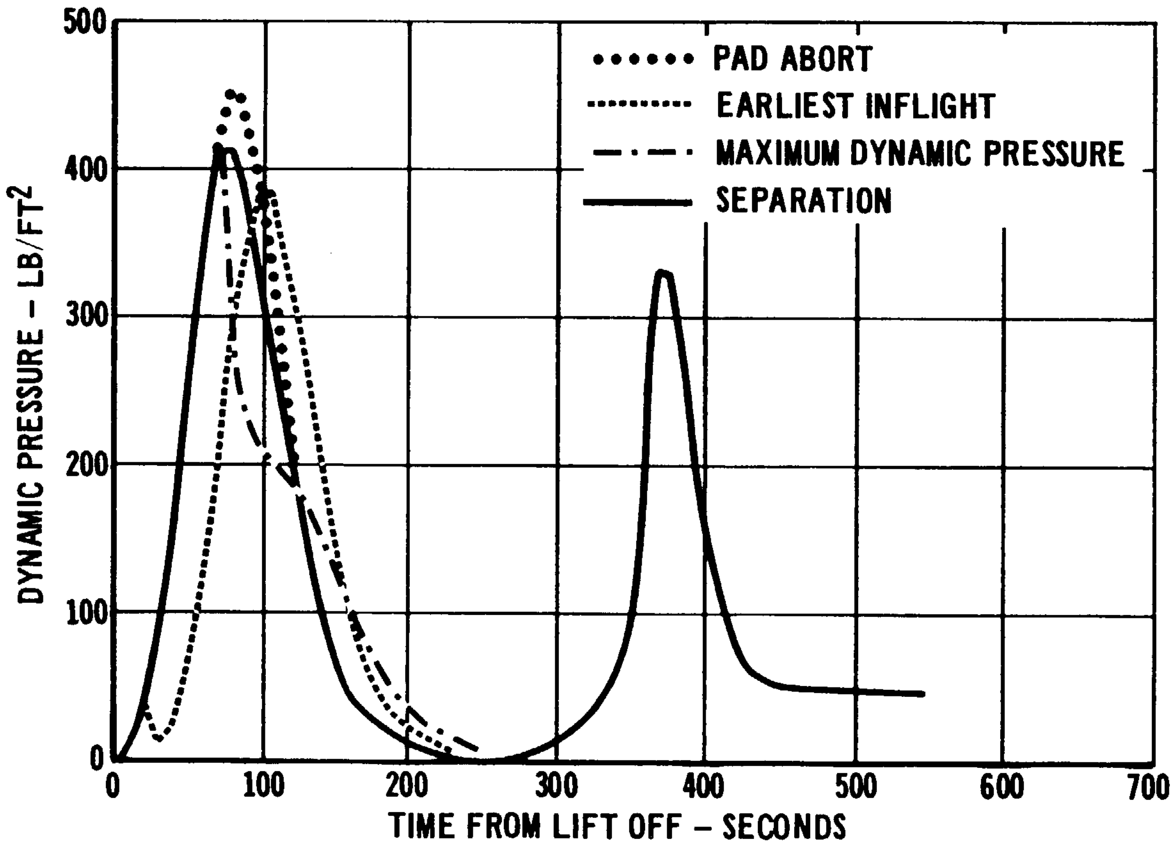


ILRVS-360F

DYNAMIC PRESSURE HISTORIES FOR ABORTS

The "nominal" abort trajectory for the HL-10 is an unpowered reentry from the nominal separation conditions. If the carrier fails before separation, the HL-10 can fly alone to nominal separation and from there safely reenter and land at Great Exuma Island or Long Island. The figure on the opposite page shows dynamic pressure histories for failures at such pre-separation times. Peak pressures are comparable in all cases.

DYNAMIC PRESSURE HISTORIES FOR ABORTS

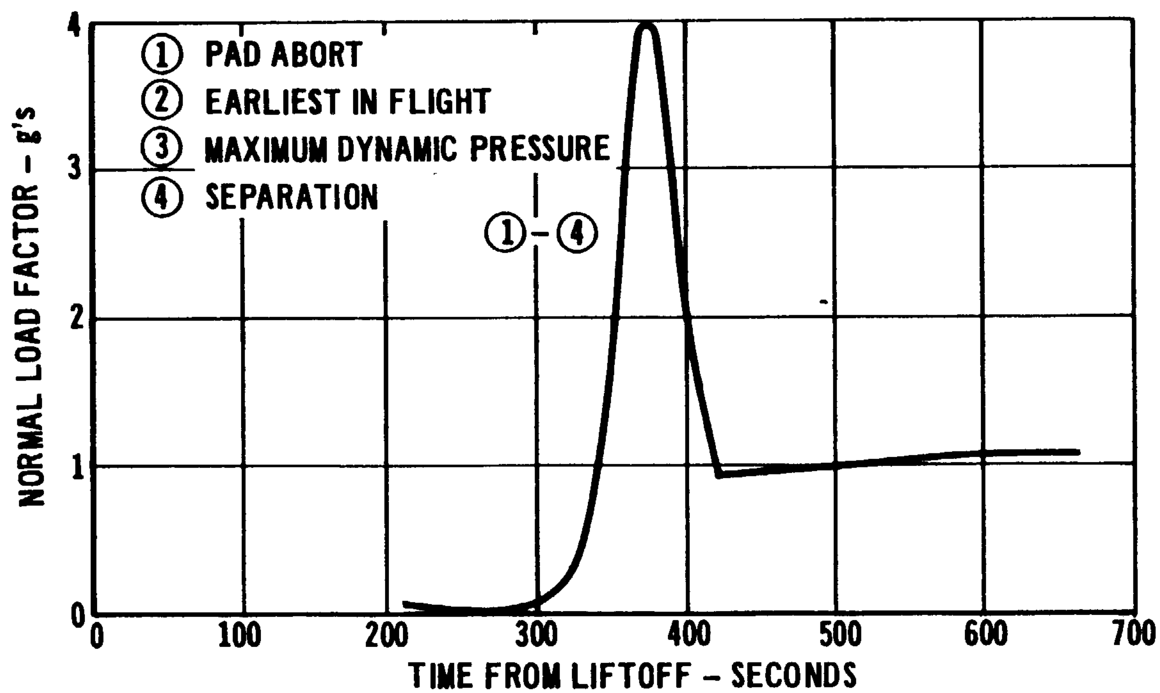


ILRVS-422F

NORMAL LOAD FACTOR HISTORIES FOR ABORTS

This figure shows the normal load factor histories corresponding to the abort trajectories discussed on the previous page. Note that for aborts the load factor has been allowed to increase to 4g's. This condition permits shortening of the range and makes landing at Great Exuma possible.

NORMAL LOAD FACTOR HISTORIES FOR ABORTS



ILRVS-423F

ABORT CONCLUSIONS

Intact abort and recovery capability will be achieved without design penalty by the use of multipe redundancy in subsystems affecting the integrity of the spacecraft during the design definition phase.

For series burn, safe abort has been found impossible during the first 20 seconds of flight. At all other times the HL-10 can at least be brought to a flyable attitude.

Possible abort landing sites exist only for certain launch azimuths when launching from Cape Kennedy. The baseline mission offers such safe landing.

ABORT CONCLUSIONS

- INTACT ABORT & RECOVERY CAPABILITY WITHOUT DESIGN PENALTY
- APPROXIMATELY 20 SECOND "DEAD-BAND"
- ALL AZIMUTH SAFE ABORT
  - ETR – NO
  - CONTINENTAL LAUNCH – YES

ILRVS-488F

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APPROACH

AND

TERMINAL LANDING

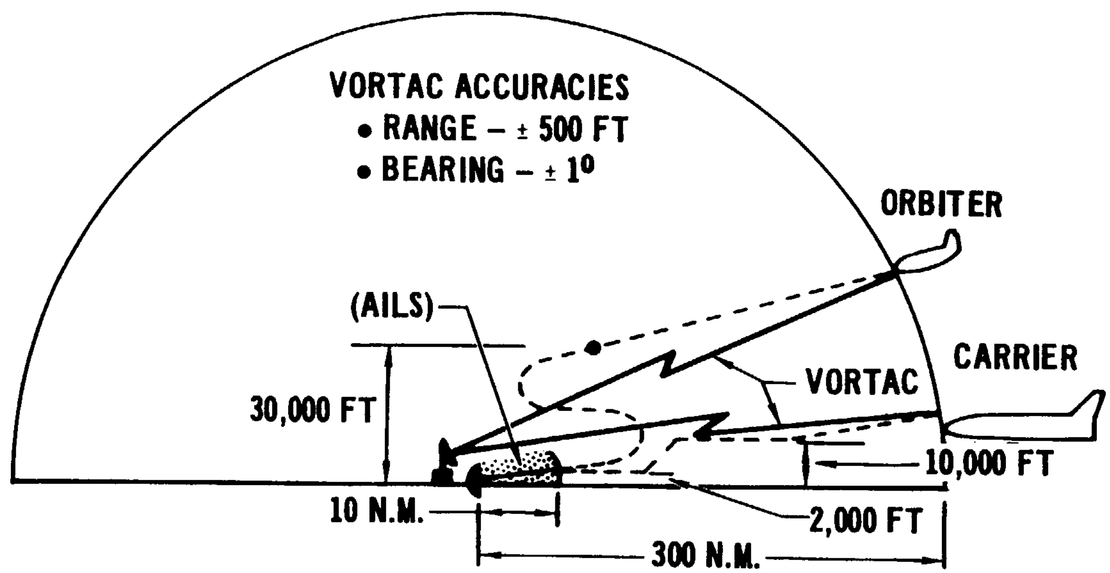
Special Emphasis Area

ILRVS-430 F

AUTOMATIC APPROACH AND LANDING SYSTEM CONCEPT

The inertial navigation system is updated from range and bearing information derived from a Vortac station located near the landing site. The improvement in navigational accuracy provided by the Vortac station permits the shuttle to fly a course which will intersect the 10 degree beam of the AILS. Final approach and landing guidance is derived from the AILS beam.

AUTOMATIC APPROACH AND LANDING SYSTEM CONCEPT



ILRVS-345F

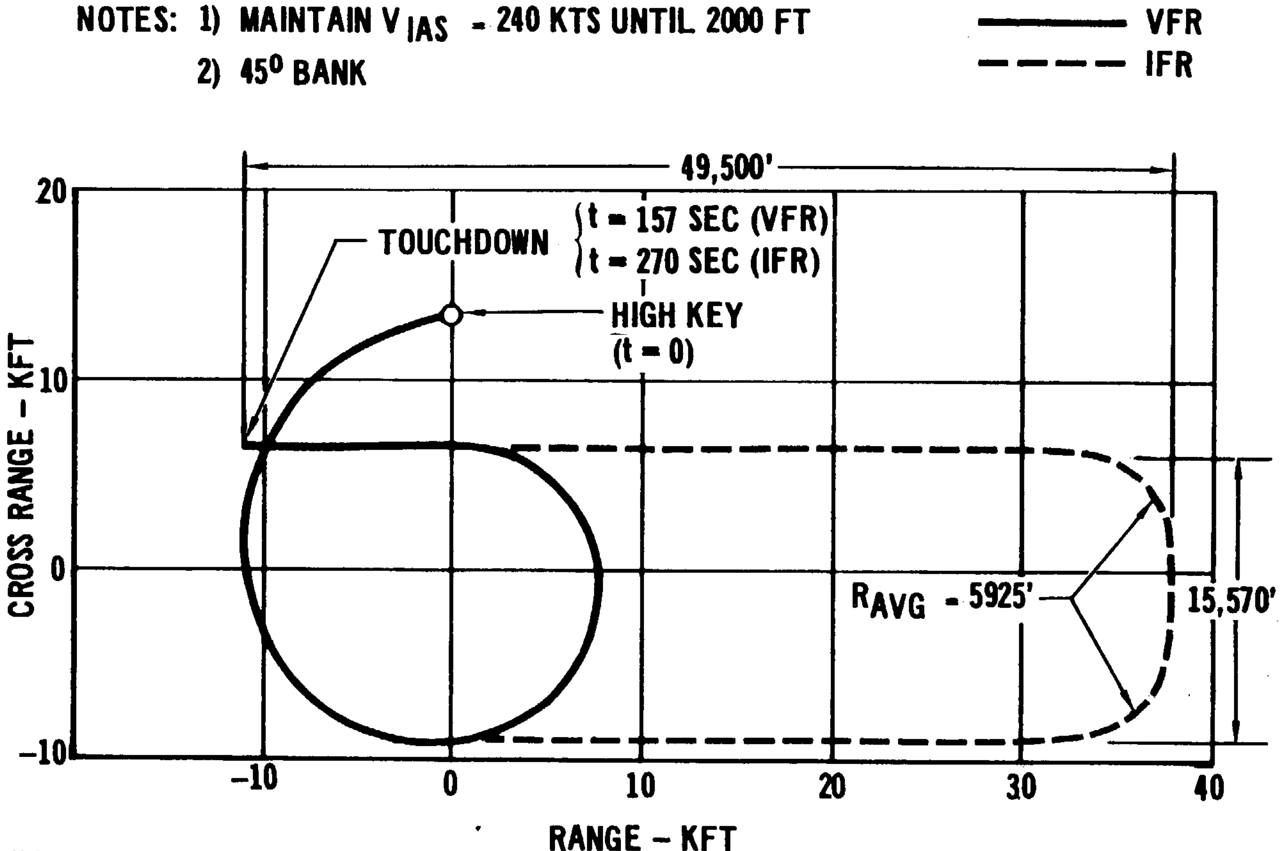
ORBITER APPROACH
(PLAN VIEW)

The planform and elevation views of typical VFR and IFR overhead approaches with a "high key" at 30,000 ft. altitude and Mach 0.644 are shown on the following two facing pages. The VFR approach shown is similar to that employed by the X-15, HL-10 and other unpowered, low L/D vehicles in landings at the Flight Research Center. During the spiral descent, the indicated airspeed and bank angle are kept constant at 240 knots and 45° respectively until roll-out for either the downwind leg of the IFR approach or the final approach of the VFR approach. The VFR approach shown is unpowered although power assist will be available from the go-around engines. The IFR approach requires power assist beginning at the turn onto the base leg ($t = 99$ sec., $h = 8000$ ft.) and continuing through the subsequent final descent which is made on a three degree glide slope at an indicated airspeed of 200 knots.

Although the approach shown requires 360 degrees of turn, this approach may be easily modified to a 270 degree approach to allow approaches from directions perpendicular to the runway. Similarly, the landing direction may be changed by 180 degrees by making the second, third and fourth (ninety degree) turns to the right instead of to the left. This maneuver will provide the capability to always land into the wind, regardless of initial approach direction.

ORBITER APPROACH
(Plan View)

NOTES: 1) MAINTAIN V_{IAS} = 240 KTS UNTIL 2000 FT
2) 45° BANK



ILRVS-453F

ORBITER APPROACH

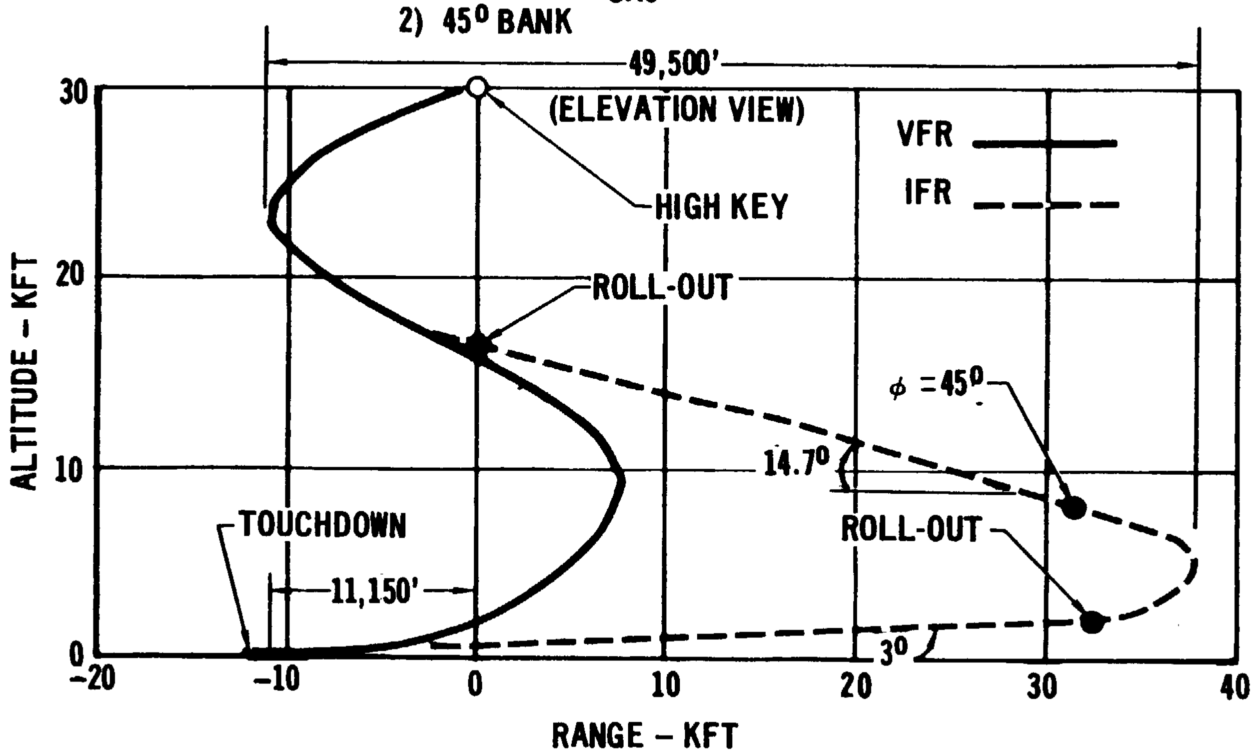
(ELEVATION VIEW)

The planform and elevation views of typical VFR and IFR overhead approaches with a "high key" at 30,000 ft. altitude and Mach 0.644 are shown on the following two facing pages. The VFR approach shown is similar to that employed by the X-15, HL-10 and other unpowered, low L/D vehicles in landings at the Flight Research Center. During the spiral descent, the indicated airspeed and bank angle are kept constant at 240 knots and 45° respectively until roll-out for either the downwind leg of the IFR approach or the final approach of the VFR approach. The VFR approach shown is unpowered although power assist will be available from the go-around engines. The IFR approach requires power assist beginning at the turn onto the base leg ($t = 99$ sec., $h = 8000$ ft.) and continuing through the subsequent final descent which is made on a three degree glide slope at an indicated airspeed of 200 knots.

Although the approach shown requires 300 degrees of turn, this approach may be easily modified to a 270 degree approach to allow approaches from directions perpendicular to the runway. Similarly, the landing direction may be changed by 180 degrees by making the second, third and fourth (ninety degree) turns to the right instead of to the left. This maneuver will provide the capability to always land into the wind, regardless of initial approach direction.

ORBITER APPROACH
(Elevation View)

NOTES: 1) MAINTAIN V_{CAS} - 240 KTS UNTIL 2000 FT
2) 45° BANK



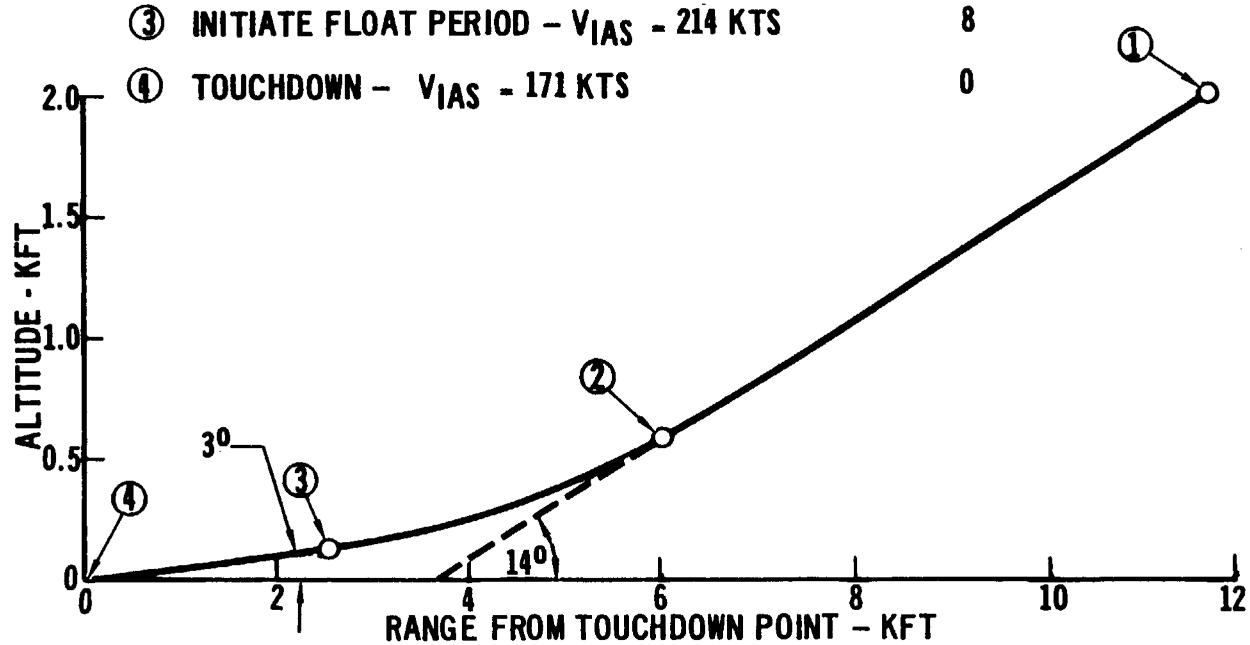
ILRVS-454F

ORBITER VFR (IDLE POWER) FINAL APPROACH

Following roll-out from the spiral descent (h = 2000 ft.), the orbiter is trimmed so that its flight path is aligned to intersect a marker at 3650 ft. from the intended touchdown point. Upon reaching an altitude of 575 ft., the landing gear is lowered and a constant 1.5g normal load factor flare is executed until a flight path angle of -3 degrees is attained. The subsequent descent is made on a 3 degree glide slope which is maintained by angle of attack modulation. Touchdown occurs at 23° angle of attack at a speed of 171 knots.

ORBITER VFR (IDLE POWER) FINAL APPROACH

EVENT	TIME FROM TOUCHDOWN (SEC)
① ROLL-OUT ON FINAL APPROACH – V_{IAS} = 240 KTS	30
② INITIATE 1.5g FLARE; LOWER LANDING GEAR	15
③ INITIATE FLOAT PERIOD – V_{IAS} = 214 KTS	8
④ TOUCHDOWN – V_{IAS} = 171 KTS	0

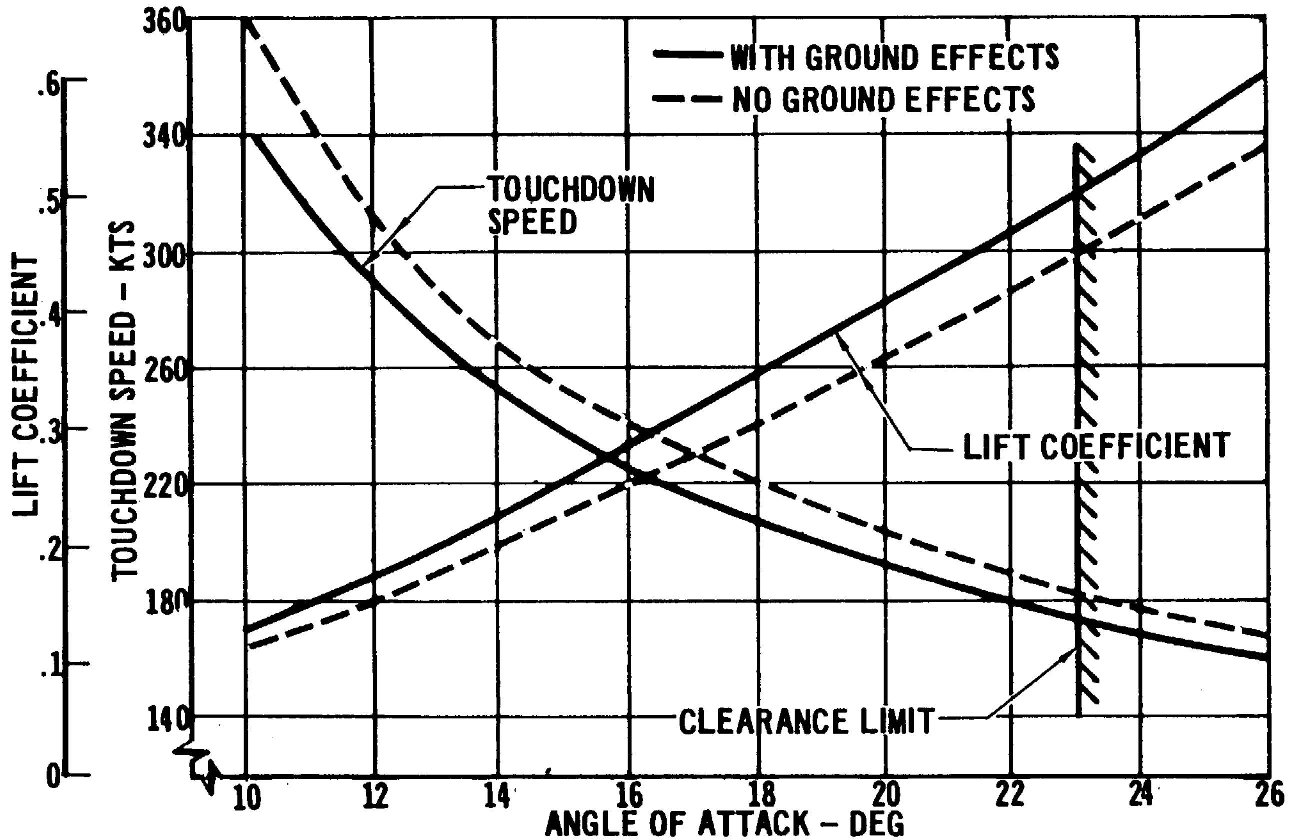


ILRVS-464F

ORBITER LANDING CHARACTERISTICS

The landing characteristics of the orbiter are shown on the facing page for a wing loading of 49.75 lb/ft^2. The estimated ground effect increases the lift curve slope by eight percent at α = 23 degrees. The corresponding touchdown speed is 171 kts.

ORBITER LANDING CHARACTERISTICS

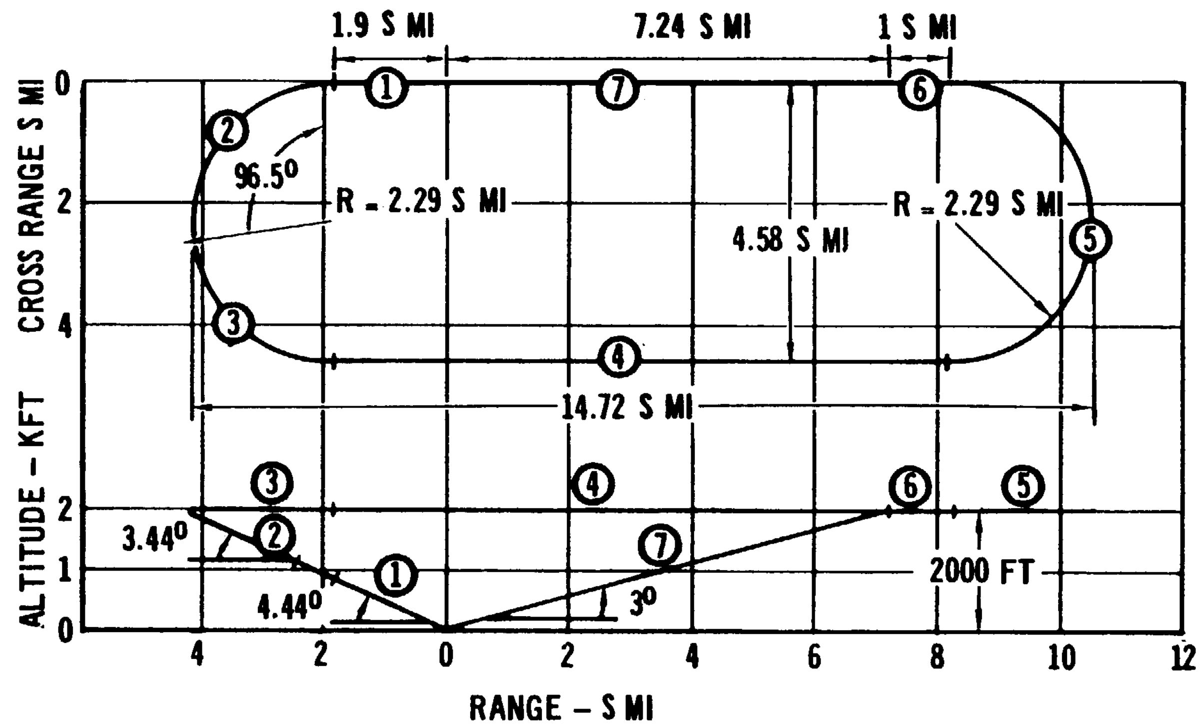


ILRVS-466F

ORBITER GO-AROUND

The orbiter go-around pattern geometry is shown on the facing page and is essentially the same as that designed for the carrier vehicle. Pattern altitude is 2000 ft. above sea level and the final descent is initiated 7.24 miles from the end of the runway at the intersection of 2000 ft. and a 3 degree glide slope. The fuel required for the IFR approach is 2000 lbs. Another 6800 lbs. are required for the go-around. This total fuel required (8800 lbs.) satisfies the amount available (9100 lbs) based upon the criterion of 3/4 engines at maximum continuous power for 10 minutes at 2000 ft. altitude.

ORBITER GO-AROUND PATTERN



ILRVS-451F

ORBITER GO-AROUND
(SEQUENCE OF EVENTS)

The major events in the orbiter go-around are tabulated on the facing page. This pattern is similar to the carrier IFR pattern. The go-around engines are sized to produce steady (thrust = drag) flight at 2000 ft. with three out of four engines operating for 10 minutes. Therefore, there is an excess thrust available (T = 1.333D) which results in a rate of climb of 1750 ft/min at sea level.

ORBITER GO-AROUND PATTERN SEQUENCE OF EVENTS

EVENT	TIME AT INITIATION OF EVENT REFERENCED TO WAVE OFF, SEC.
1 CLIMB OUT AT $\dot{h} = 29.1$ FT/SEC; $\gamma = 4.44^0$	0
2 20^0 BANKED TURN; $\dot{h} = 22.6$ FT/SEC; $\gamma = 3.44^0$	27
3 LEVEL FLIGHT 20^0 BANKED TURN	80
4 CRUISE AT $h = 2{,}000$ FT – $V_{IAS} = 218$ KTS	127
5 LEVEL FLIGHT 20^0 BANKED TURN	243
6 CONSTANT ALTITUDE DECELERATION	344
7 DESCEND ON 3^0 GLIDE SLOPE; $\dot{h} = -17.7$ FT/SEC	359
8 TOUCHDOWN AT 171 KTS	472

ILRVS–452F

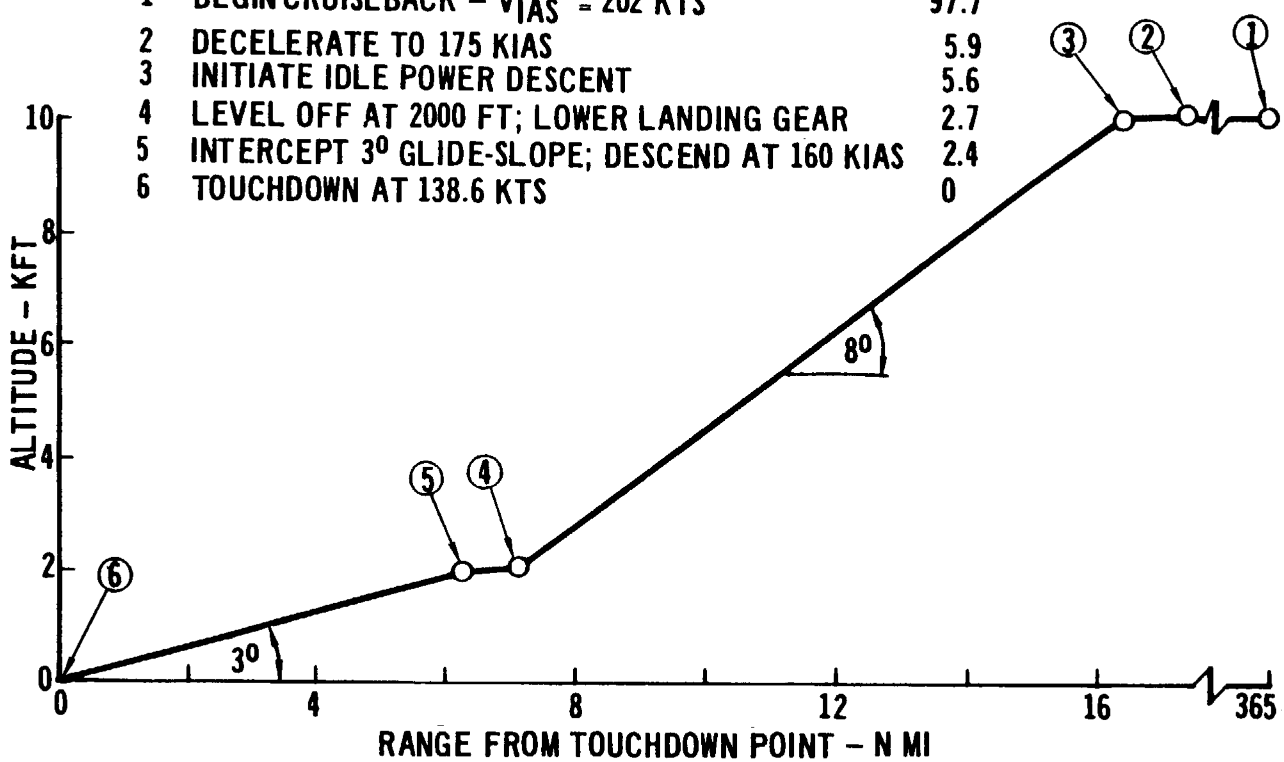
CARRIER IFR APPROACH

Cruiseback attitude is 10,000 ft. and Mach $\simeq$ 0.3. When the carrier is 17.4 na. mi. (twenty statute miles) from the landing site, the pilot decelerates to 175 KIAS and reduces power to set up an 8° glide slope until an altitude of 2000 ft is reached at which time he levels off, lowers the landing gear and decelerates to 160 knts. The final approach is made on a 3° glider slope using power to maintain V_{IAS} = 160 kts until near the ground when the final flare is made to reduce the rate of sink.

The VFR approach is essentially made in the same manner with the exception that the final approach may be initiated somewhat closer to the runway and the approach descent made at a higher rate of sink.

CARRIER IFR APPROACH

	EVENT	TIME FROM TOUCHDOWN (MIN)
1	BEGIN CRUISEBACK – V_{IAS} = 202 KTS	97.7
2	DECELERATE TO 175 KIAS	5.9
3	INITIATE IDLE POWER DESCENT	5.6
4	LEVEL OFF AT 2000 FT; LOWER LANDING GEAR	2.7
5	INTERCEPT 3⁰ GLIDE-SLOPE; DESCEND AT 160 KIAS	2.4
6	TOUCHDOWN AT 138.6 KTS	0

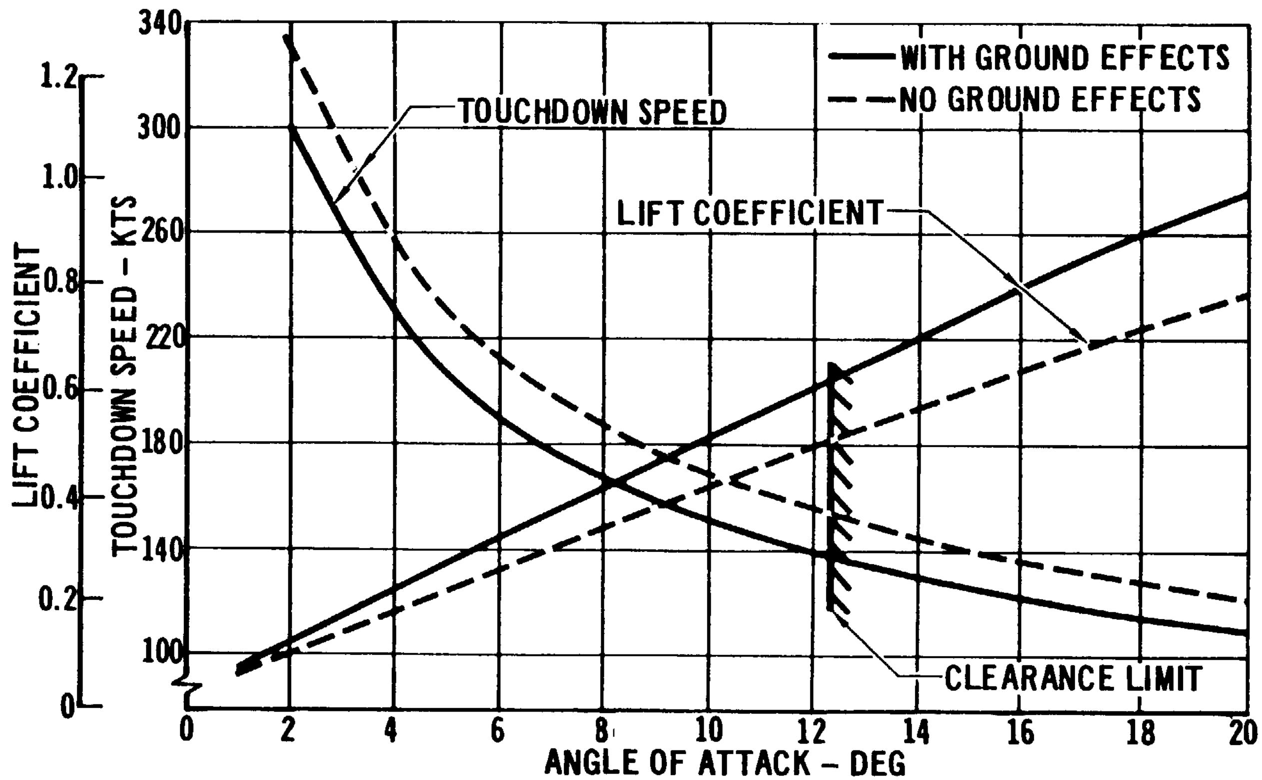


ILRVS–463F

CARRIER LANDING CHARACTERISTICS

The variation of lift coefficient with angle of attack of the carrier landing configuration and the corresponding touchdown speed variation is shown for a wing loading of 40.5 lb/ft^2 with and without ground effect at sea level. The ground effect is estimated to produce a twenty-five percent increase in the lift curve slope at an angle of attack of 12 degrees, resulting in a design touchdown speed of 138.6 kts.

CARRIER LANDING CHARACTERISTICS

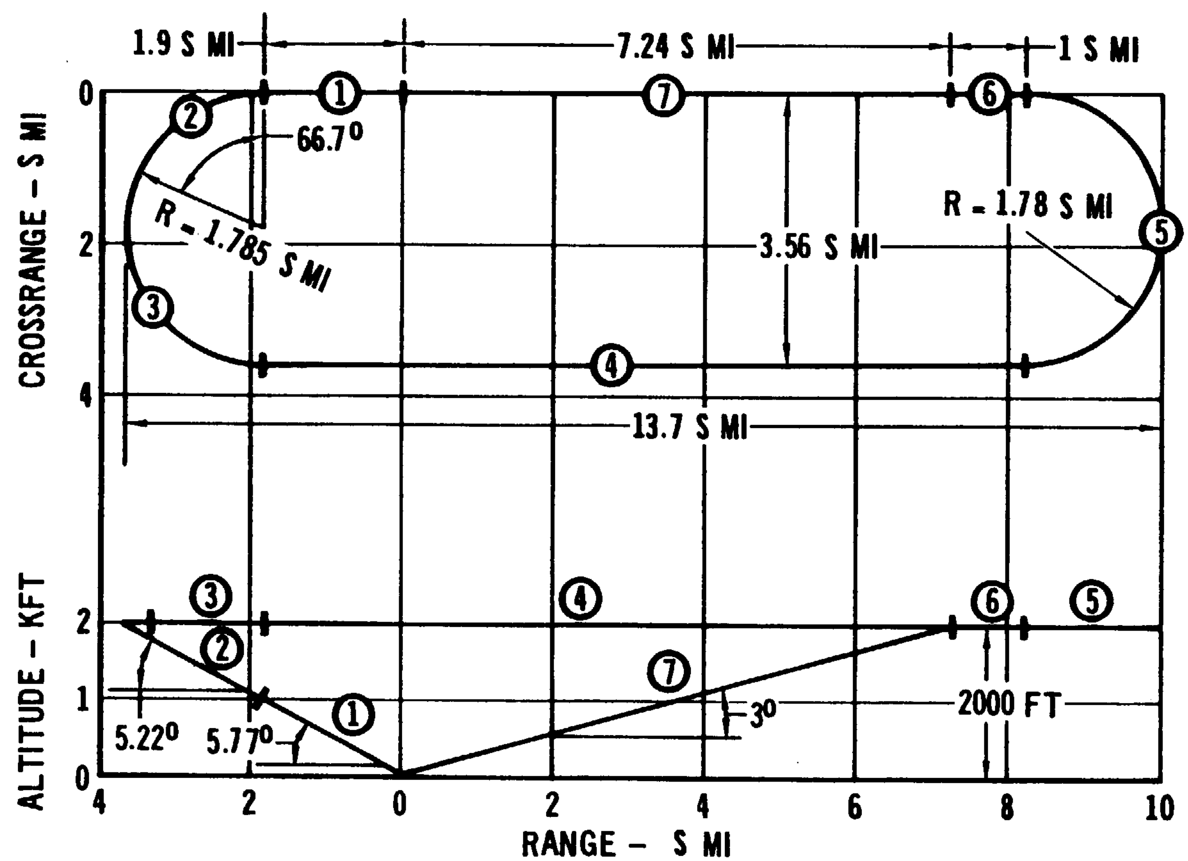


ILRVS-465F

CARRIER GO-AROUND

Following a wave-off, climb-out is made with gear up and at nearly the maximum power setting until directly over the end of the runway, at which time a 20 degree banked climbing turn to the left is made until the 2000 ft. pattern altitude is reached. The turn is continued at constant altitude until onto the downwind leg. Approximately 8 miles beyond the end of the runway, a continuous 180 degree turn onto the final approach is made, at which time the gear is lowered and the carrier decelerated so that it intercepts a 3 degree glide slope approximately 7.25 miles from the end of the runway. The final descent is made on this glide slope until touchdown. Total fuel required for the carrier go-around is 4000 lbs.

CARRIER GO-AROUND PATTERN



ILRVS-450F

CARRIER GO-AROUND
(SEQUENCE OF EVENTS)

A tabulation of the major events in the carrier go-around pattern is shown on the facing page. Times are referenced to wave-off which is assumed to occur immediately over the end of a 10,000 ft. runway. This pattern is designed to satisfy IFR conditions. Under VFR conditions, the pilot may shorten the pattern length considerably.

CARRIER GO-AROUND PATTERN
SEQUENCE OF EVENTS

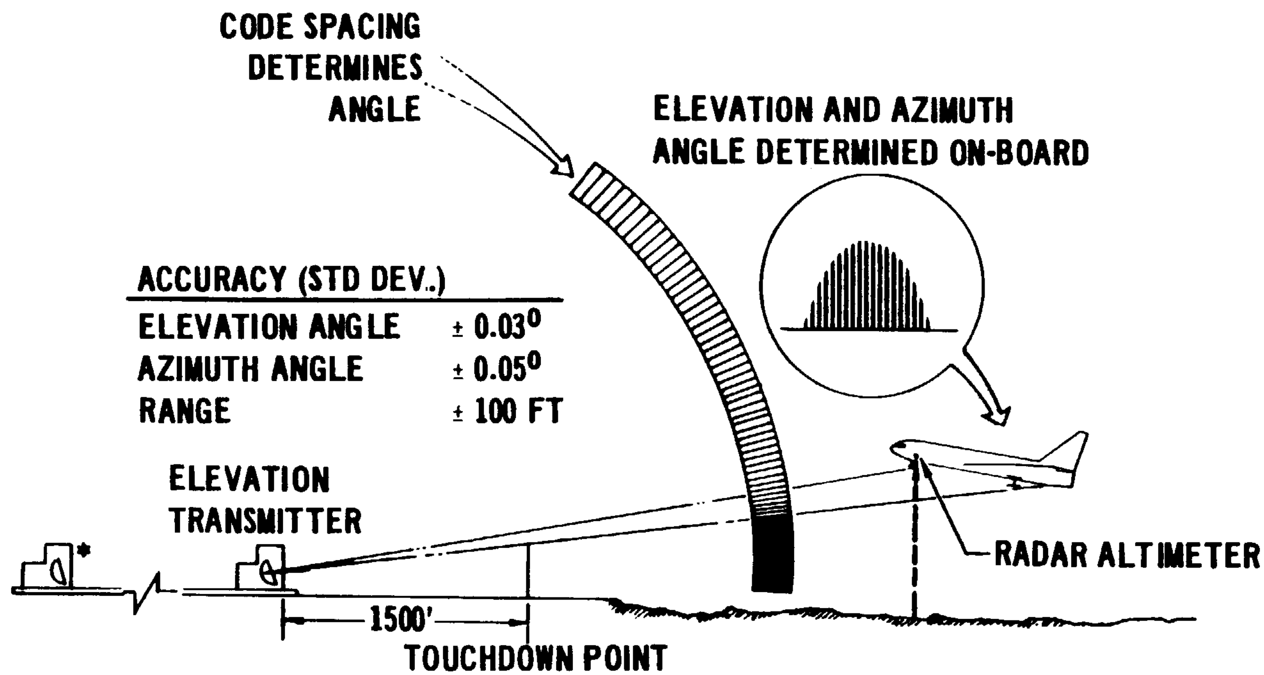
EVENT	TIME AT INITIATION OF EVENT REFERENCED TO WAVE OFF, SEC.
① CLIMB OUT AT $\dot{h} = 33.3$ FT/SEC; $\gamma = 5.77^{\circ}$	0
② 20° BANKED TURN; $\dot{h} = 30.2$ FT/SEC; $\gamma = 5.22^{\circ}$	30
③ LEVEL FLIGHT 20° BANKED TURN	63
④ CRUISE AT $h = 2{,}000$ FT – $V_{IAS} = 191$ KTS	119
⑤ LEVEL FLIGHT 20° BANKED TURN	280
⑥ CONSTANT ALTITUDE DECELERATION	369
⑦ DESCEND ON 3° GLIDE SLOPE; $\dot{h} = 15.5$ FT/SEC	386
⑧ TOUCHDOWN AT 138.6 KTS	514

ILRVS–455F

ADVANCED INSTRUMENT LANDING SYSTEM

The AILS ground transmitter provides a coded microwave scanning beam which permits on-board detection of elevation and azimuth relative to the landing strip. Range to the end of runway or touchdown point is also provided. AILS provides information which allows automatic landing including roll-out. The AILS uses the same guidance principals as does present-day ILS. The AILS hardware design has improved the guidance accuracy and eliminated beam errors caused by overflying aircraft and ground clutter. Other systems such as the SPN-42 radar are capable of providing automatic landing but the AILS is less complex, similar to present ILS and is expected to be certified by the F.A.A.

ADVANCED INSTRUMENT LANDING SYSTEM
Elevation Technique Shown-Azimuth Concept Identical



* AZIMUTH & RANGE TRANSMITTER LOCATED AT ROLL-OUT END OF RUNWAY

ILRVS-328F

AUTOMATIC POWERED LANDING - HL-10

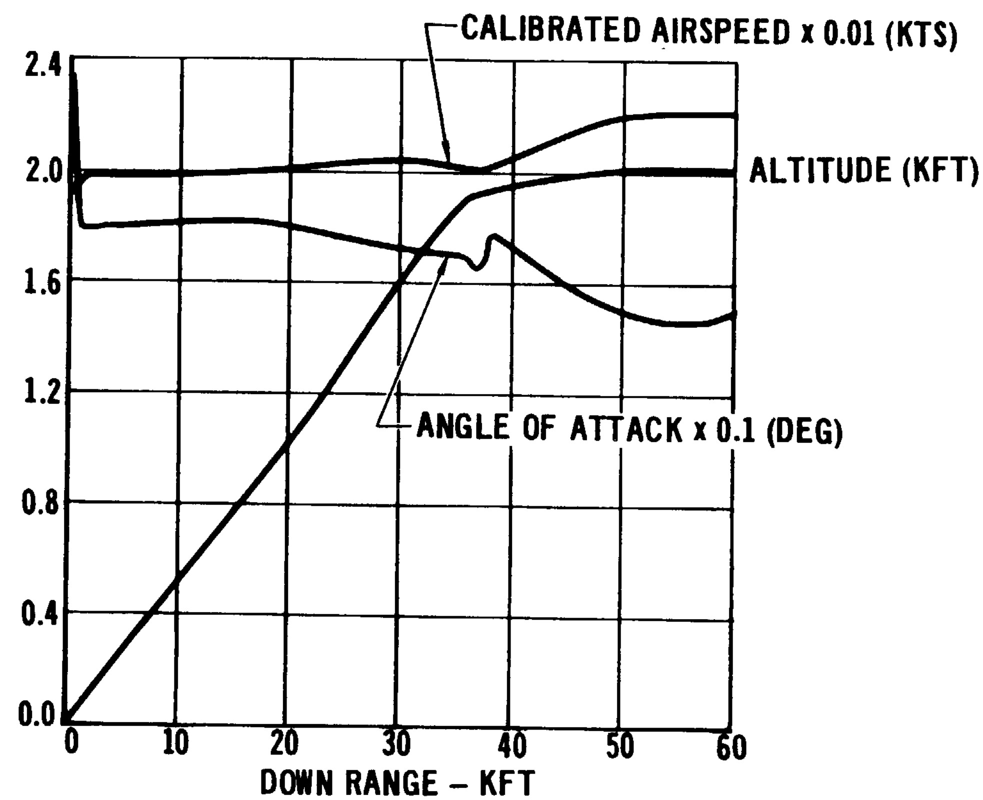
A digital computer program generated the landing trajectory shown. The assummed events prior to this simulation are:

1. The engines have been deployed and are operating
2. The vehicle has been aligned to the runway at the appropriate speed and altitude in a level attitude.

During the landing trajectory, the velocity is lowered to the desired glide slope velocity followed by the acquisition of the glide slope. The transients settle out prior to the flare maneuver.

The feasibility of this technique, using comparatively simple standard types of equipment, was demcnstrated.

AUTOMATIC POWERED LANDING
HL-10



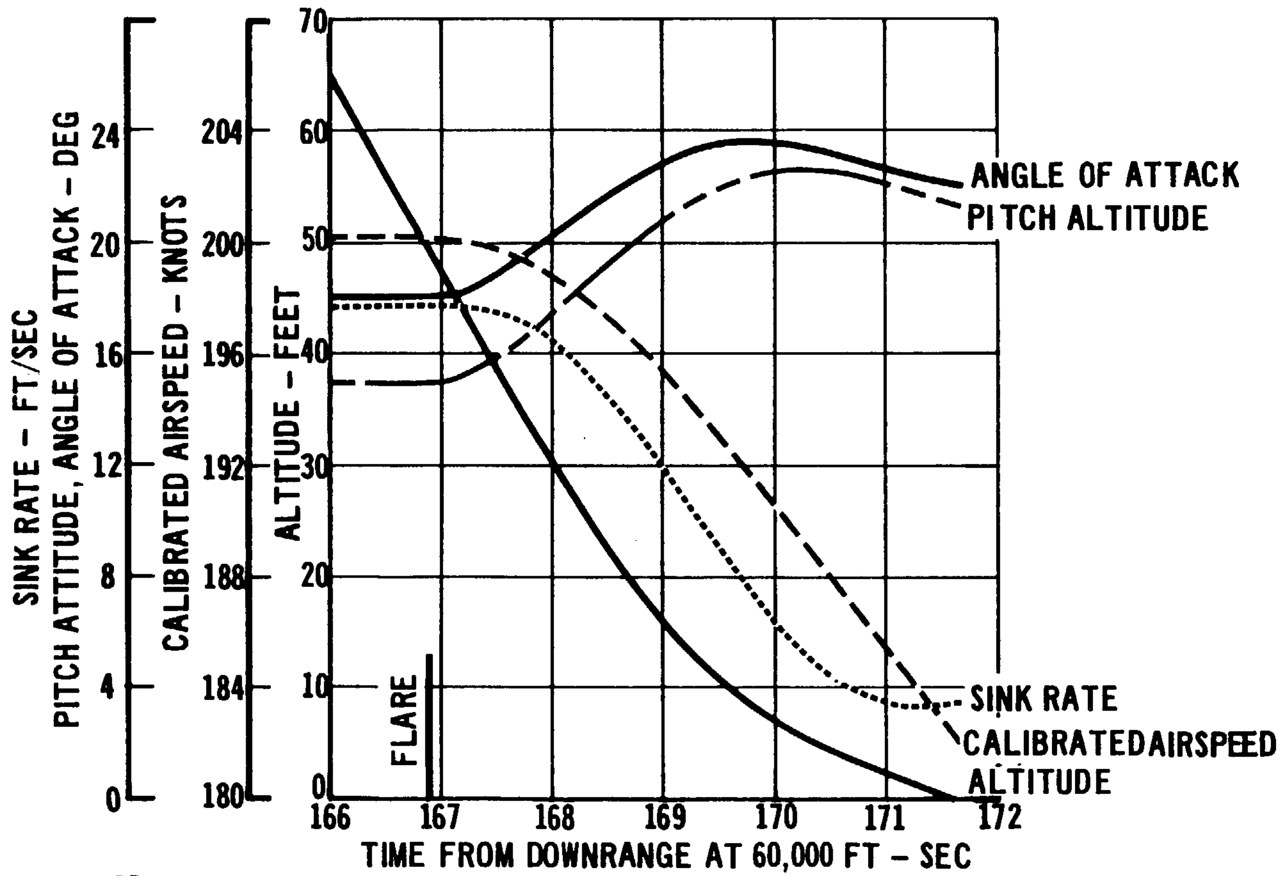
ILRVS-346F

AUTOMATIC POWERED LANDING
HL-10 FLARE DETAILS

From a stable powered glide condition, the flare maneuver is initiated. At an altitude of 50 feet, flare commences with the shutdown of the engine, the insertion of an open loop pitch up bias, and the use of a different set of control gains.

While other flare techniques may be satisfactory, this one gives satisfactory touchdown conditions of a sink rate of less than 4 feet/second and a velocity of 182 knots.

AUTOMATIC POWERED LANDING

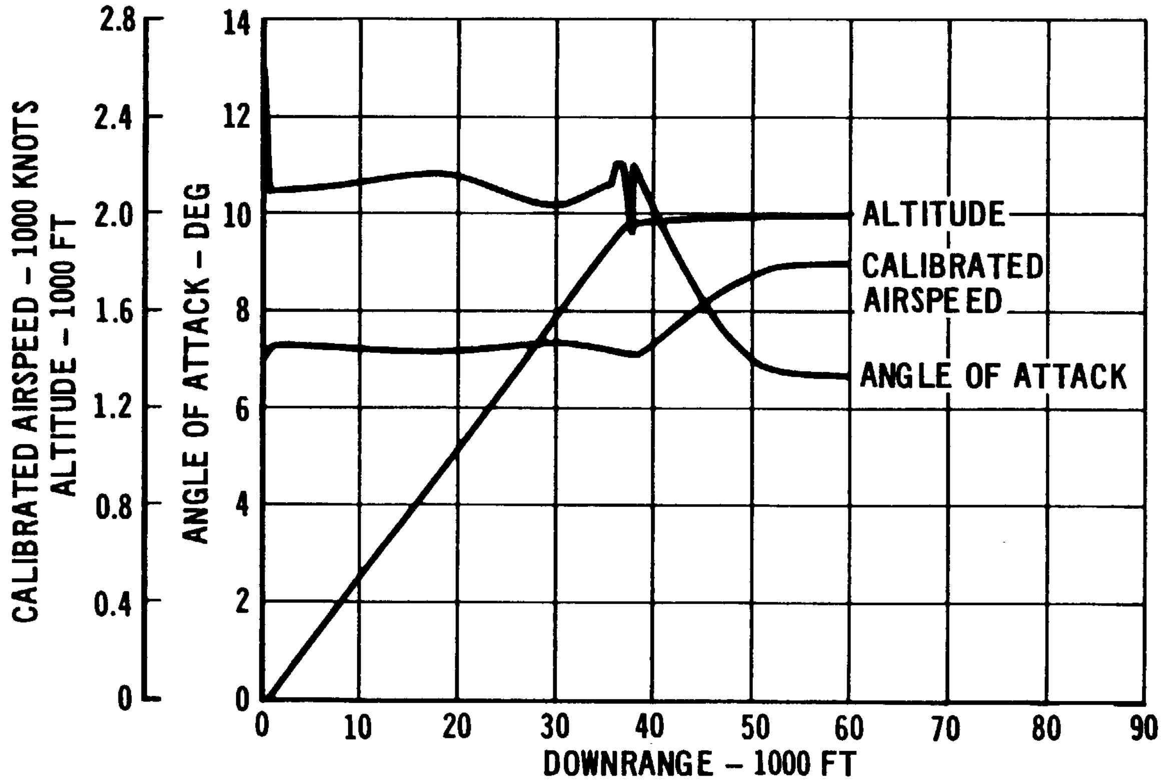


ILRVS-347F

AUTOMATIC POWERED LANDING - CARRIER

The sequence of events is the same as for the HL-10 but the initial and terminal conditions are different. The same glide slope is utilized.

POWERED AUTOMATIC LANDING CARRIER NOMINAL TRAJECTORY

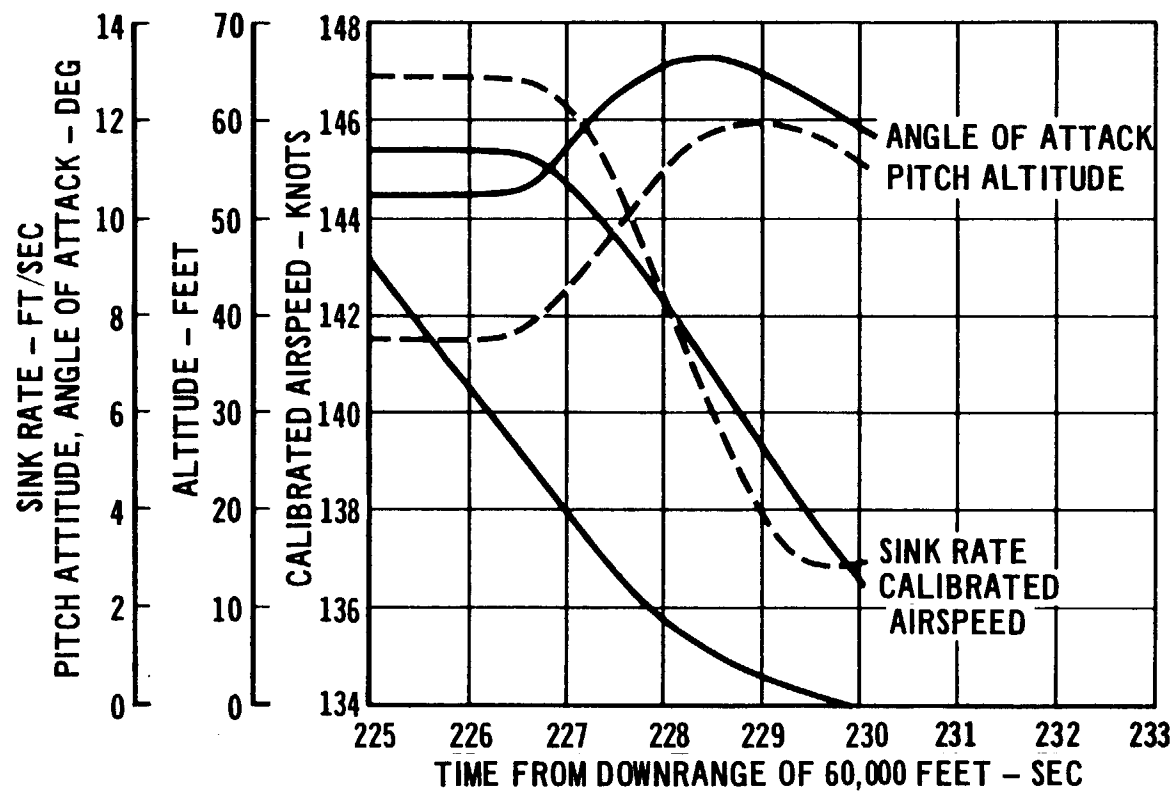


ILRVS-492F

AUTOMATIC POWERED LANDING
CARRIER FLARE DETAILS

The flare is initiated at 30 feet, but the technique is otherwise the same as for the orbiter. The pitch attitude is less than 12 degrees at touchdown.

POWERED AUTOMATIC LANDING
CARRIER FLARE DETAILS



ILRVS-491F

AUTOMATIC LANDING EQUIPMENT REQUIREMENTS

The equipment required to perform a satisfactory landing is within the capabilities of currently available equipment. The air data instruments are conventional types. The radar altimeter is a low range type of about 500 feet maximum range. The glide slope determination is not required for a range greater than 7 nautical miles since the approach altitude is 2000 feet and a 3 degree glide slope is used. The autopilot and flare electronics are part of the existing computer. The high altitude data could be supplied by an updated inertial reference instead of an air instrument.

AUTOMATIC LANDING EQUIPMENT REQUIREMENTS

- AIR DATA
  - AIRSPEED (± 2%)
  - ALTITUDE (OR UPDATED INERTIAL REFERENCE) (± 2%)
- LOW ALTITUDE RADAR ALTIMETER (± 2 FT)
- PRECISE GROUND BASED GLIDE SLOPE DETERMINATION (AILS OR PRECISE RADAR)
- AUTOPILOT
- FLARE ELECTRONICS

ILRVS-449F

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INTEGRATED

ELECTRONICS

SYSTEMS

Special Emphasis Area

ILRVS-440F

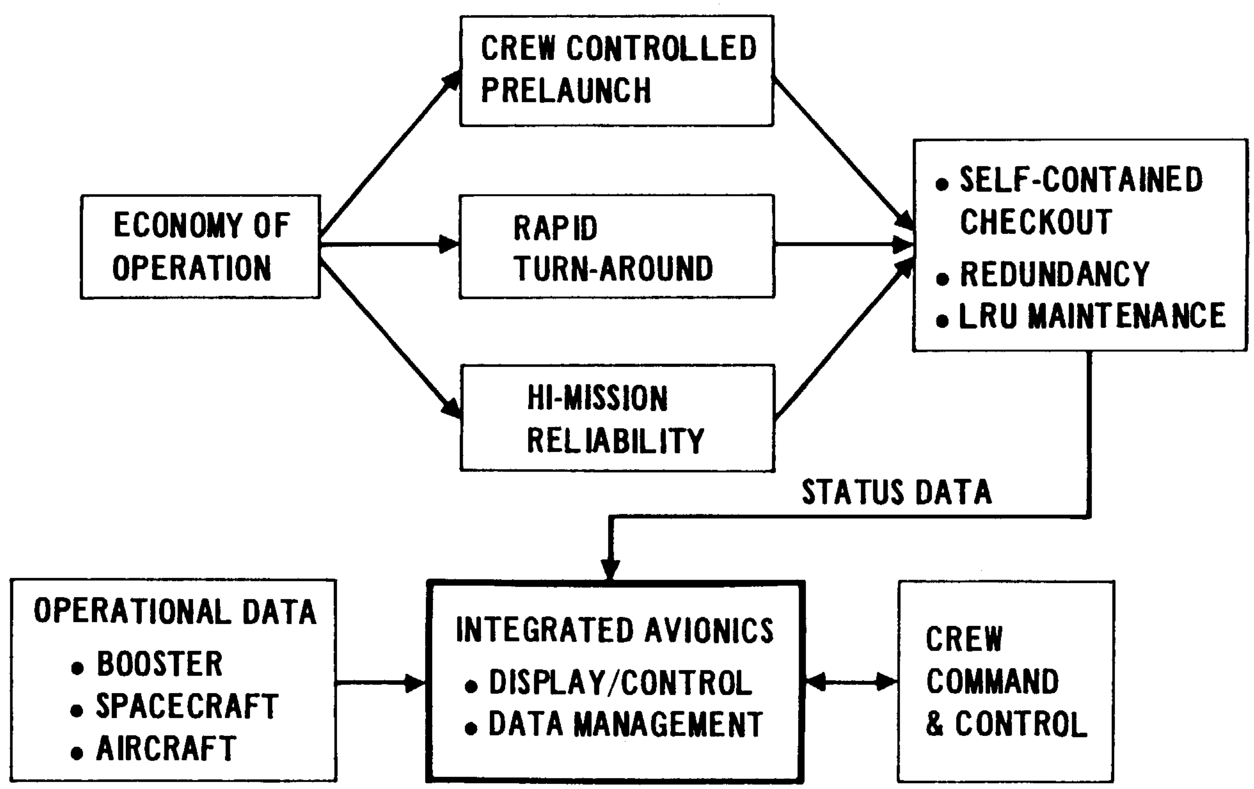
7

RATIONALE FOR INTEGRATED AVIONICS

The primary reason for integrated avionics is to meet the goal of operational economy. This economy can be met by achieving autonomous prelaunch capability, reuse with rapid turnaround and high mission success. Key avionic techniques to achieve these capabilities are self checkout, redundancy and line replaceable unit maintenance. This, however, increases the data to be handled. Vehicle operation over many flight regimes also increases the equipment/sensor complexity and data.

An integrated avionics system with a high degree of sophistication is thus required to insure efficient processing of this large quantity of system data while simultaneously provided the crew with command/controllability.

RATIONALE FOR INTEGRATED AVIONICS



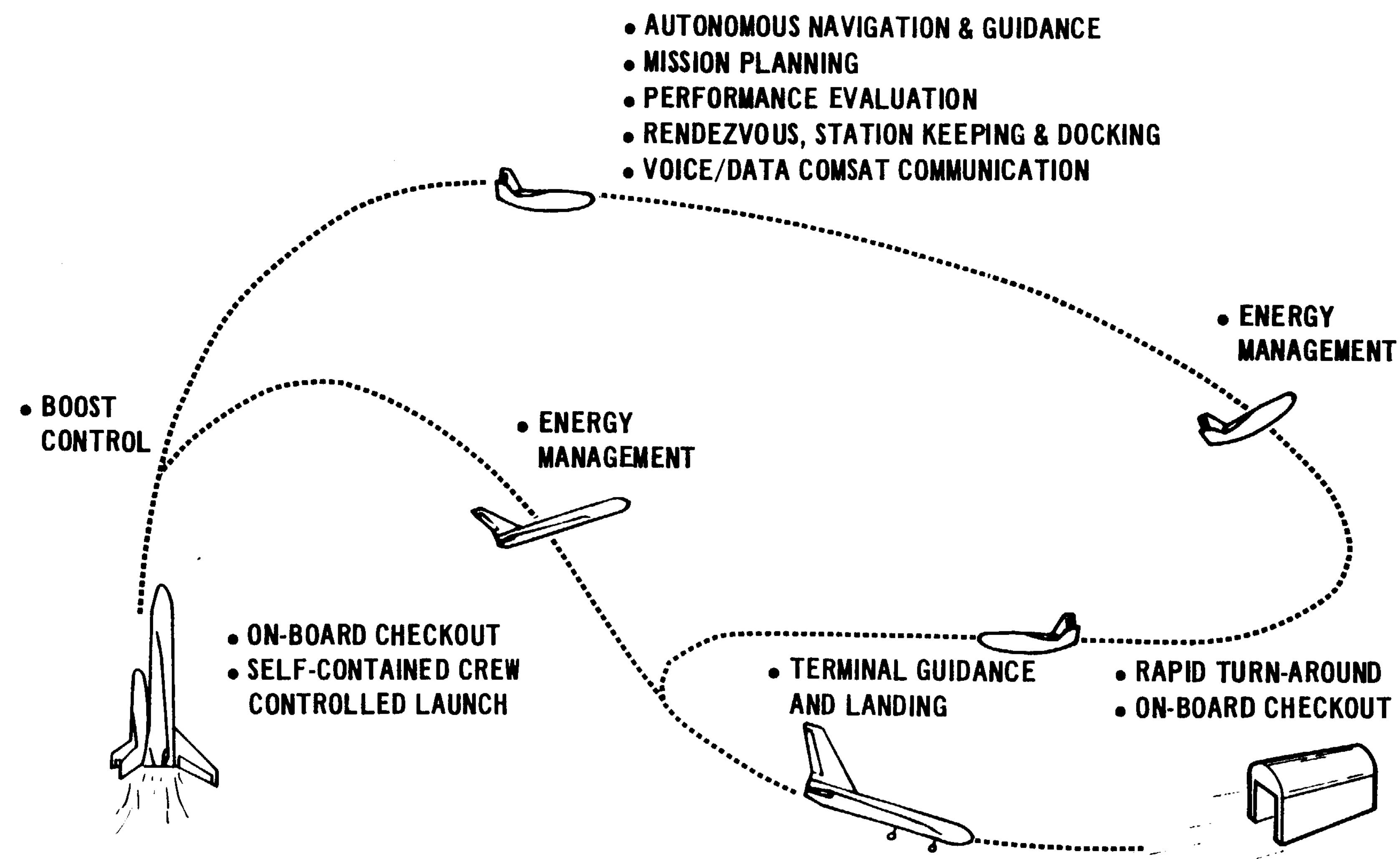
ILRVS-285F

AVIONICS-MISSION FUNCTIONS

This chart depicts avionic functions by mission phase. Key requirements are to provide economy of operation using:

- o Crew controlled prelaunch checkout
- o On orbit autonomous navigation, rendezvous, mission planning and evaluation. Trend data for ground maintenance
- o On orbit ComSAT communications
- o Terminal guidance and all weather landing capability
- o Post landing - status evaluation and quick maintenance for reuse.

AVIONICS - MISSION FUNCTIONS



ILRVS-286r

KEY CONCEPTS AND TRADEOFFS

This chart lists the major driving concepts behind the avionics system design configuration. Each concept has options or alternate methods of implementation. The tradeoff decisions and selection rationale will be presented on succeeding charts.

KEY CONCEPTS AND TRADE OFFS

DATA MANAGEMENT	• DEGREE OF COMPUTATIONAL DISTRIBUTION (CENTRALIZED VS DECENTRALIZED) • DATA INTERFACE TECHNIQUE (MULTIPLEXED VS NONMULTIPLEXED)
ON-BOARD CHECKOUT	• SEPARATE CHECKOUT SYSTEM VS DECENTRALIZED BUILT-IN TEST • MANUAL VS AUTOMATIC • LEVEL OF FAULT ISOLATION AND MAINTENANCE
DISPLAY & CONTROL	• MULTIMODE INTEGRATED DISPLAYS VS SINGLE PURPOSE INDIVIDUAL DISPLAYS • UTILIZE SPECIAL HEADS UP DISPLAY
RELIABILITY & REUSE	• USE OF BLOCK VS FUNCTIONAL REDUNDANCY • INTERACTION OF REDUNDANCY VS SELF TEST • EVALUATION OF ACTIVE/PASSIVE/STANDBY REDUNDANCY • MALFUNCTION DETECTION AND SWITCHOVER

ILRVS-287 F

COMPUTATIONAL REQUIREMENTS

The data management system is involved with the total complement of hardware and software utilized for acquisition, processing, analysis and distribution of information to the crew and other using subsystems. This listing of subsystem information/computational requirements indicates the magnitude of the data management task.

In addition to conventional spacecraft computations such as guidance and navigation, the space shuttle has unique requirements such as propulsion performance trend data analysis for purposes of expediting ground maintenance.

COMPUTATIONAL REQUIREMENTS

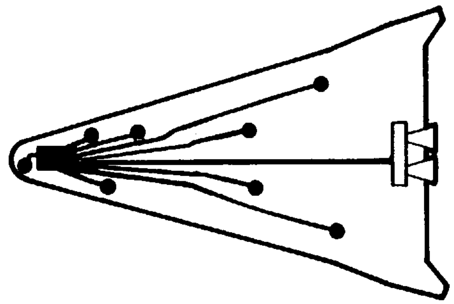
GUIDANCE	ASCENT, ORBIT, RENDEZVOUS, REENTRY, LANDING, ABORT
NAVIGATION	INERTIAL, AUGMENTED INERTIAL, AUTONOMOUS
FLIGHT CONTROL	ATTITUDE, STABILIZATION
ON-BOARD MISSION PLANNING	TRAJECTORY GENERATION, OPTIMIZATION & SELECTION, FLIGHT PROGRAM IDENTIFICATION, LOAD ALLEVIATION, ACCESSMENT OF UPLINK INFORMATION, CREW USAGE FOR SCIENTIFIC CALCULATIONS
CONFIGURATION AND SEQUENCING CONTROL	PAYLOAD PREPARATION & DEPLOYMENT, SYSTEM READINESS, SENSOR CONTROL, SAFING OPERATIONS, EXPERIMENT ACTIVATION & CONTROL, PILOT CHECK LIST
CREW DISPLAYS	SYMBOL GENERATION – PRIORITY & FUNCTIONAL SORTING
ON-BOARD CHECKOUT	STIMULUS GENERATION, PARAMETER TOLERANCE BAND COMPARISON, TREND DATA EVALUATION
PROPULSION	OPERATION, PROPELLANT UTILIZATION, MALFUNCTION DETECTION, TREND ANALYSIS
BIOMEDICAL	INFORMATION PROCESSING & EVALUATION – HEART RATE, BREATH RATE, FLUIDS ANALYSIS
DATA BUS MANAGEMENT	REQUEST/REPLY OPERATION, MESSAGE TRANSFER VERIFICATION

ILRVS-288 F

DATA MANAGEMENT SYSTEM DISTRIBUTION

The centralization versus decentralization of computational equipment is a major consideration in determining the design philosophy and subsequent design configuration of the data management system. A conceptual trade study of five alternate computational approaches was performed. The hybrid approach was selected as the baseline system configuration.

DATA MANAGEMENT SYSTEM DISTRIBUTION

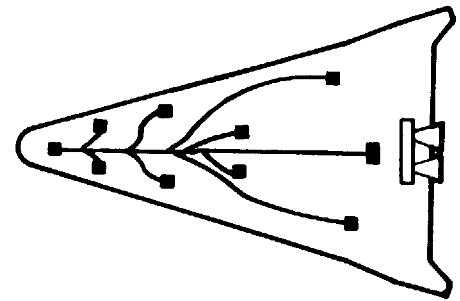


CENTRALIZED – COMPUTER/MULTIPROCESSOR

- LARGE COMPUTER REQUIREMENTS
- MAXIMUM DATA TRANSFER & BUS REQUIREMENTS
- MULTIPROCESSORS UNDEVELOPED
- COMPLEX SOFTWARE

COMPUTATIONAL COMMONALITY

- GROUP SIMILAR CALCULATIONS INTO FEWER COMPUTING UNITS
- MINIMIZE DATA BUSSES AND WIRES

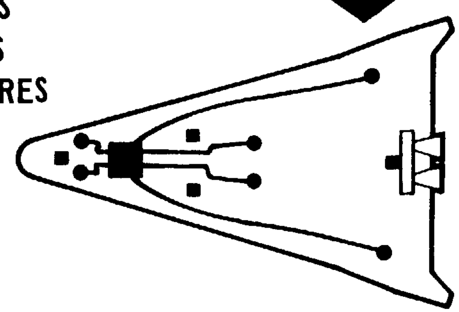


DECENTRALIZED – SUBSYSTEM DEDICATED COMPUTERS

- 30 COMPUTERS REQUIRED (ORBITER)
- COMPLEX EXECUTIVE CONTROL/INTERFACE
- MANY DESIGN SPECS VENDORS

PHYSICAL LOCATION COMMONALITY

- EQUIPMENT LOCATION IMPACTS DATA TRANSFER



HYBRID APPROACH – COMMONALITY OF FUNCTION AND LOCATION

- MINIMUM NUMBER SPECIAL PURPOSE COMPUTERS (SENSOR ORIENTED)
- ONE GENERAL PURPOSECENTRAL COMPUTER (MISSION ORIENTED)
- GREATER DESIGN AND MISSION FLEXIBILITY

ILRVS–289 F

COMPUTER SYSTEM

This block diagram shows the inter-relationship of the assemblies and identifies the major signal interfaces with other vehicle subsystems.

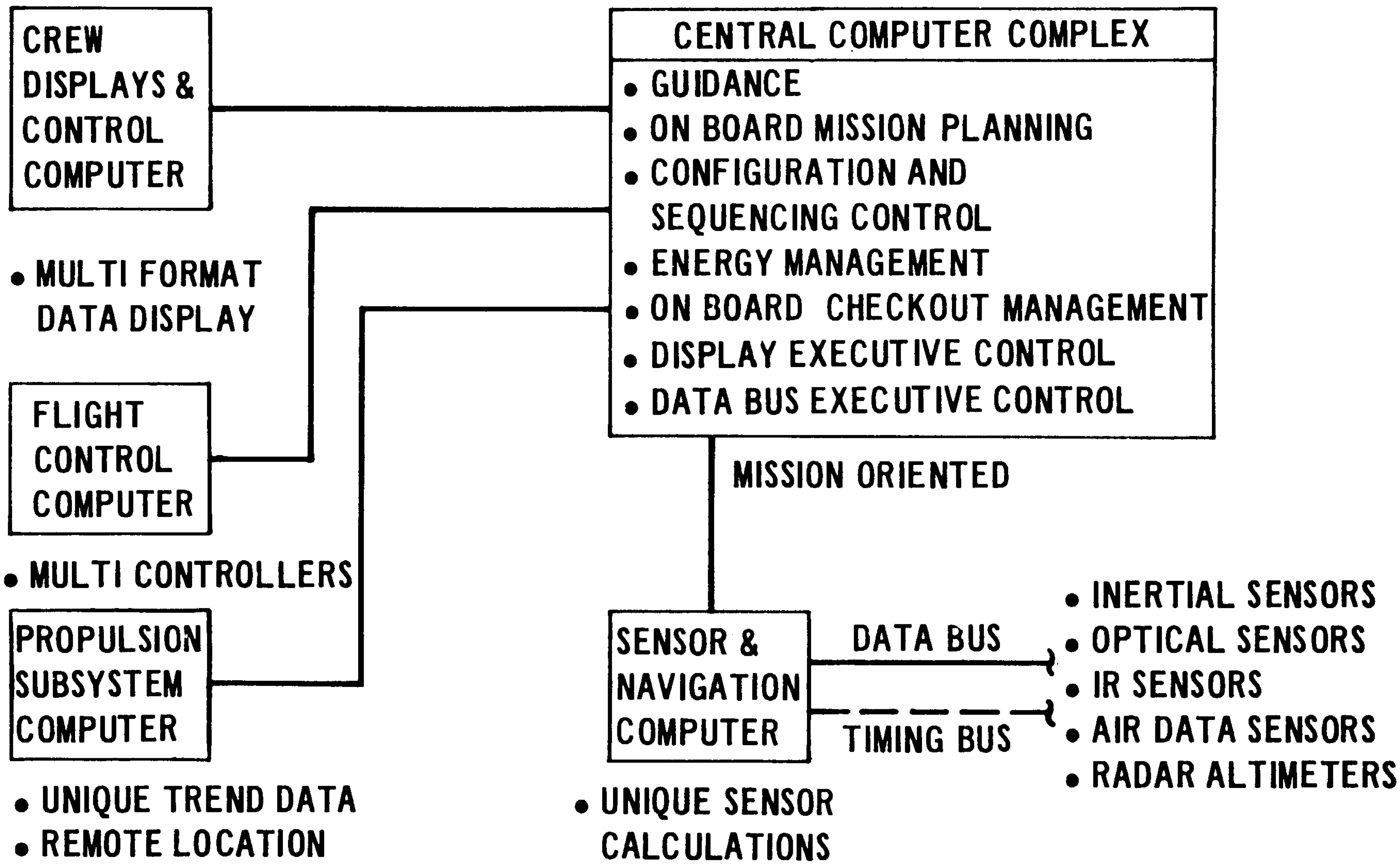
The selected allocation of computers consists of a central computer complex performing mission oriented functions and peripheral dedicated computers for sensor oriented functions and was chosen on the basis of commonality of requirements and physical location.

As an example of the advantage of grouping like computations in the central computer, the guidance algorithms may be used for both guidance and mission trajectory planning computations. The software can be modularized to reduce costs and provide software redundancy.

Special purpose dedicated computers are assigned to the following subsystems for the reasons indicated.

- o Crew displays - Large symbology memory storage; high speed calculations for CRT multi data formats.
- o Sensor & Navigation - High iteration rate and unique type computations such as strapdown IMU coordinate determinations.
- o Flight Control - Large amounts of data associated with many mission modes, e.g., thrusters, aerodynamic surfaces, brakes.
- o Propulsion - Many, remotely located engines; large amount of data for calculations such as propellant utilization.

COMPUTER SYSTEM



ILRVS-290F

ON-BOARD SELF TEST AND CHECKOUT

The prime purpose of the on-board checkout system on the space shuttle is to provide autonomous operation and rapid turnaround time with improved probability of mission success. The implementation techniques to achieve these goals are indicated on this chart.

ON-BOARD SELF TEST AND CHECKOUT

- PROVIDES CREW CONTROLLED PRELAUNCH AND LAUNCH CAPABILITY
  - SELF TEST
  - CONTINUOUS EVALUATION
  - STATUS DISPLAY
  - MONITOR OF ALL VEHICLE SUBSYSTEMS
- PROVIDES RAPID TURN AROUND CAPABILITY
  - BUILT IN FAULT IDENTIFICATION TO LINE REPLACEABLE UNIT (LRU)
  - RECORDED TREND DATA
  - REDUCED TROUBLE SHOOTING
- IMPROVED PROBABILITY OF MISSION SUCCESS
  - CONTINUOUS FAILURE DETECTION
  - AUTOMATIC SWITCHING TO REDUNDANT UNITS

ILRVS-291F

ON BOARD CHECKOUT SELECTION

The decentralized built in test approach was selected because it minimizes interface complexity and it provides subsystem functional autonomy. Test stimuli and response measuring equipment are included in each line replaceable unit. This eliminates the need for external test stimuli and external special purpose test circuitry required with the separate centralized test system. Since the BIT is independent of central computer operation the ground maintenance crew can perform maintenance to the go-no-go level without use of a computer or other test equipment.

ON BOARD CHECKOUT SELECTION

APPROACH:

√ • DECENTRALIZED BUILT IN TEST

• SEPARATE SYSTEM

RATIONALE:

• MINIMIZE INTERFACE COMPLEXITY

• SUBSYSTEM FUNCTIONAL AUTONOMY

√ SELECTED

ILRVS–490F

ON-BOARD SELF TEST AND CHECKOUT SYSTEM

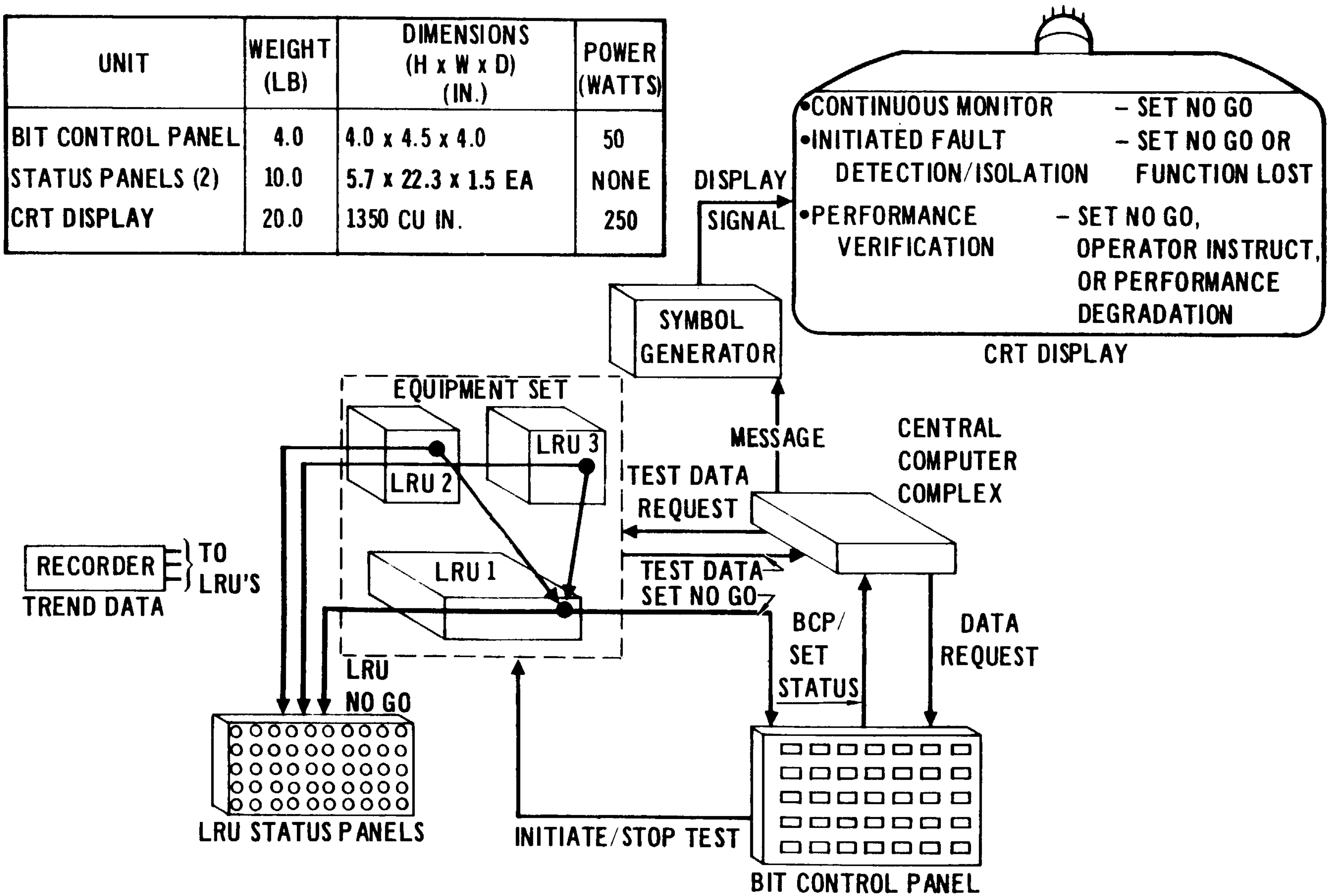
Faults are indicated on the BIT control panel by the illumination of the corresponding set lamps, via discrete signal lines independent of central computer or operator action. When the Central Computer Complex (CCC) detects a failed set indication on the BIT control panel, the faulty set name is transferred from the CCC to the symbol generator to pilot display. Also, latching fault indicators on the LRU status panel and on the LRU are activated, via discrete signal lines independent of central computer or operator action, for those failures detected in an individual LRU by continuous monitor BIT.

Expanded fault detection/isolation or diagnostic test routines, of the equipment sets, can be manually initiated from the BIT control panel via the initiate/stop test line. The computer receives the equipment set test results and formats the message. The message is then transferred from the CCC to the symbol generator to the display for viewing by the pilot or ground maintenance technician. Fault indications resulting from the fault detection/isolation or diagnostic tests are also indicated on the LRU and LRU status panel latching indicators. This status readout is accomplished without use of the CCC.

LRU trend/diagnostic data is stored by a magnetic tape recorder. The recorded data is used by the ground crew for impending failure predictions to expedite maintenance.

ON-BOARD SELF TEST AND CHECKOUT SYSTEM

UNIT	WEIGHT (LB)	DIMENSIONS (H x W x D) (IN.)	POWER (WATTS)
BIT CONTROL PANEL	4.0	4.0 x 4.5 x 4.0	50
STATUS PANELS (2)	10.0	5.7 x 22.3 x 1.5 EA	NONE
CRT DISPLAY	20.0	1350 CU IN.	250

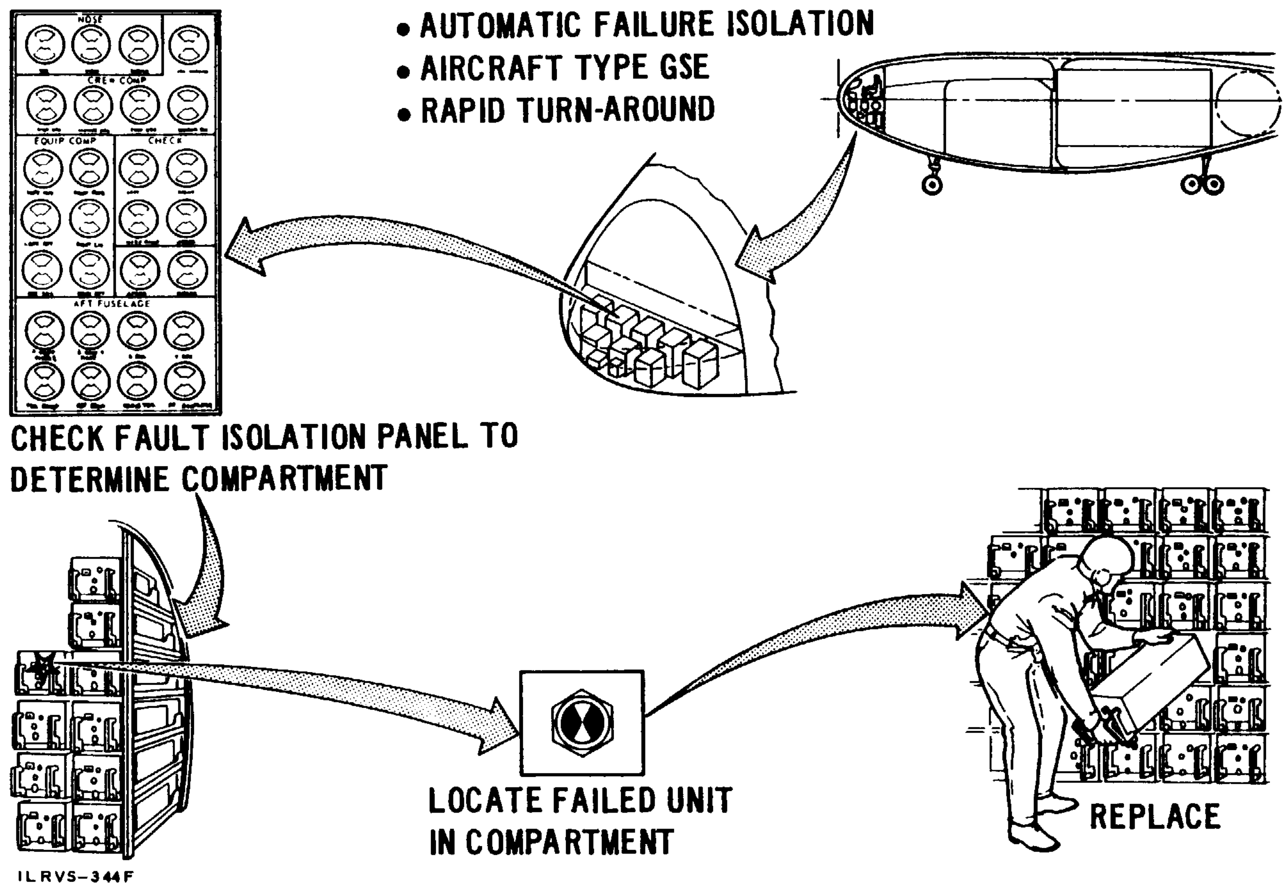


GROUND MAINTENANCE CONCEPT

A centrally and easily accessible status panel provides a latching indication of the compartment in which a LRU has failed. This indication of failure is used by the post-flight maintenance crew to direct them to the location of the failure. Once the location of failure is determined the latching indicators on the individual LRU's are used for determining the faulty unit. The latching indicators retain their status after power off and must be manually reset.

GROUND MAINTENANCE CONCEPT

- AUTOMATIC FAILURE ISOLATION
- AIRCRAFT TYPE GSE
- RAPID TURN-AROUND



OCS IMPLEMENTATION PENALTY

The BIT weight, cost and volume penalties associated with various fault detection levels, differ from subsystem to subsystem. But with all subsystems, very high fault detection levels can be achieved for relatively small penalties. For this reason, 95% to 98% levels of fault detection were selected for the space shuttle.

OCS IMPLEMENTATION PENALTY

- DEPENDENT ON DEGREE OF FAULT DETECTION CAPABILITY
- STUDIES INDICATE DIFFERENCES FROM SYSTEM TO SYSTEM

CONSIDERATION	SYSTEM	FAULT DETECTION LEVEL			
		70%	85%	95%	98%
Δ WEIGHT	CONTROLS & DISPLAYS	3%	5%	7%	10%–50 LB
	FLIGHT CONTROLS	1%	1.5%	2%	2.5%–10 LB
	COMPUTER	<< 1%	<< 1%	< 1%	< 1%– 2 LB
	NAVIGATION	4%	5%	7%	15%–100 LB
Δ COST	ALL	APPROXIMATELY SAME AS Δ WEIGHT			
Δ VOLUME	ALL	SMALL IMPACT			

ILRVS-292F

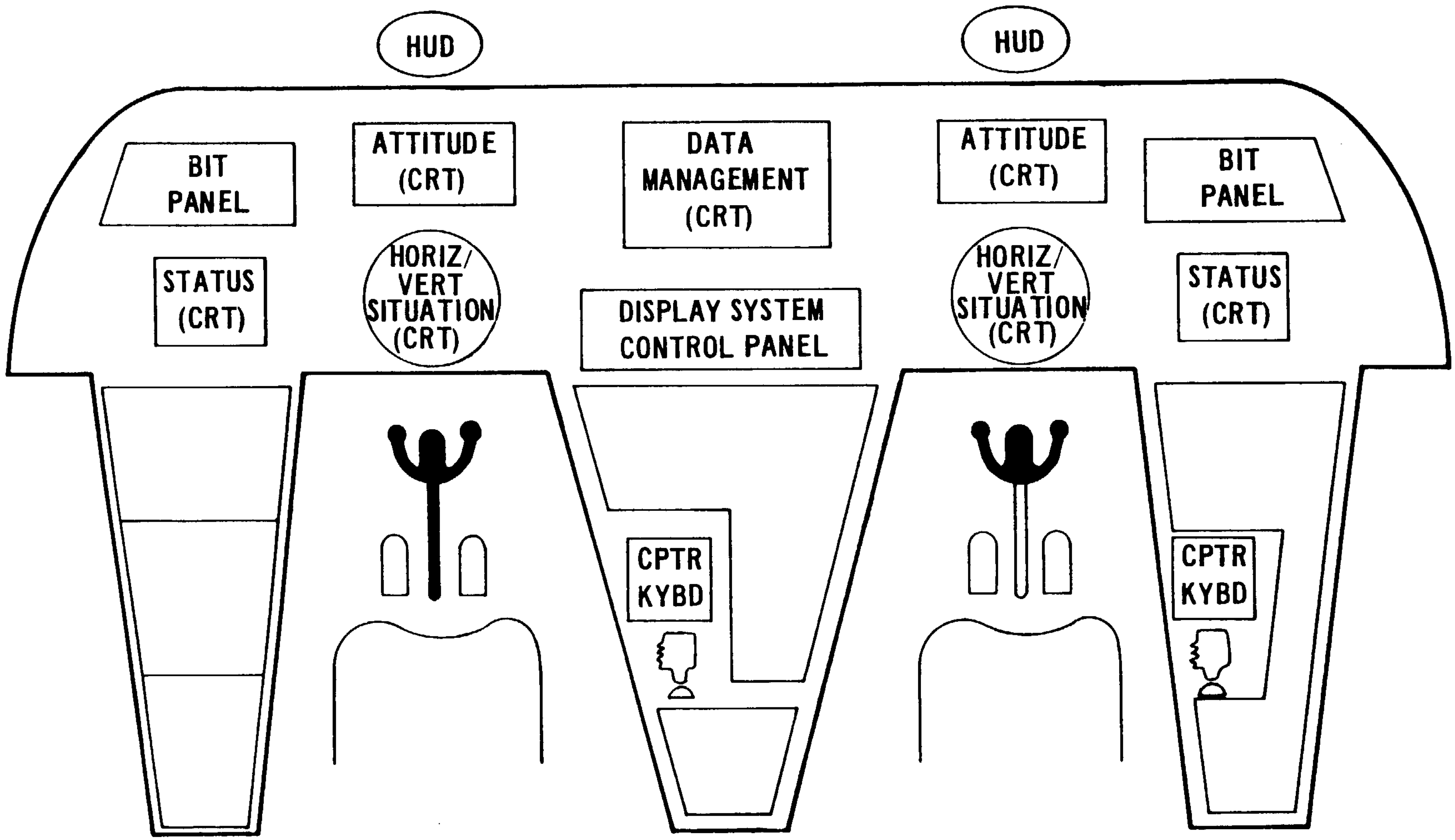
MULTIMODE DISPLAYS

The display and control layout indicates the use of cathode ray tubes (CRT) having the capability to superimpose slide data. This increases the types and quantity of data that can be displayed.

Failure and status data is automatically presented on the status CRT. However, the BIT panel is used to initiate a more complete test with results also shows on the status CRT.

A control yoke and rudder pedals are used, for control when operating as an aircraft. When operating as a spacecraft a hand controller is used. Current studies may lead to use of only hand controllers with outputs switchable to the appropriate system. Pertinent studies are MDAC F-4 and the Cornell University variable stability fly by wire test programs.

MULTI-MODE DISPLAYS



CRT AND CRT WITH SUPERIMPOSED SLIDE CAPABILITY
HEAD-UP DISPLAY (HUD) FOR STATION APPROACH AND LANDING

ILRVS-333F

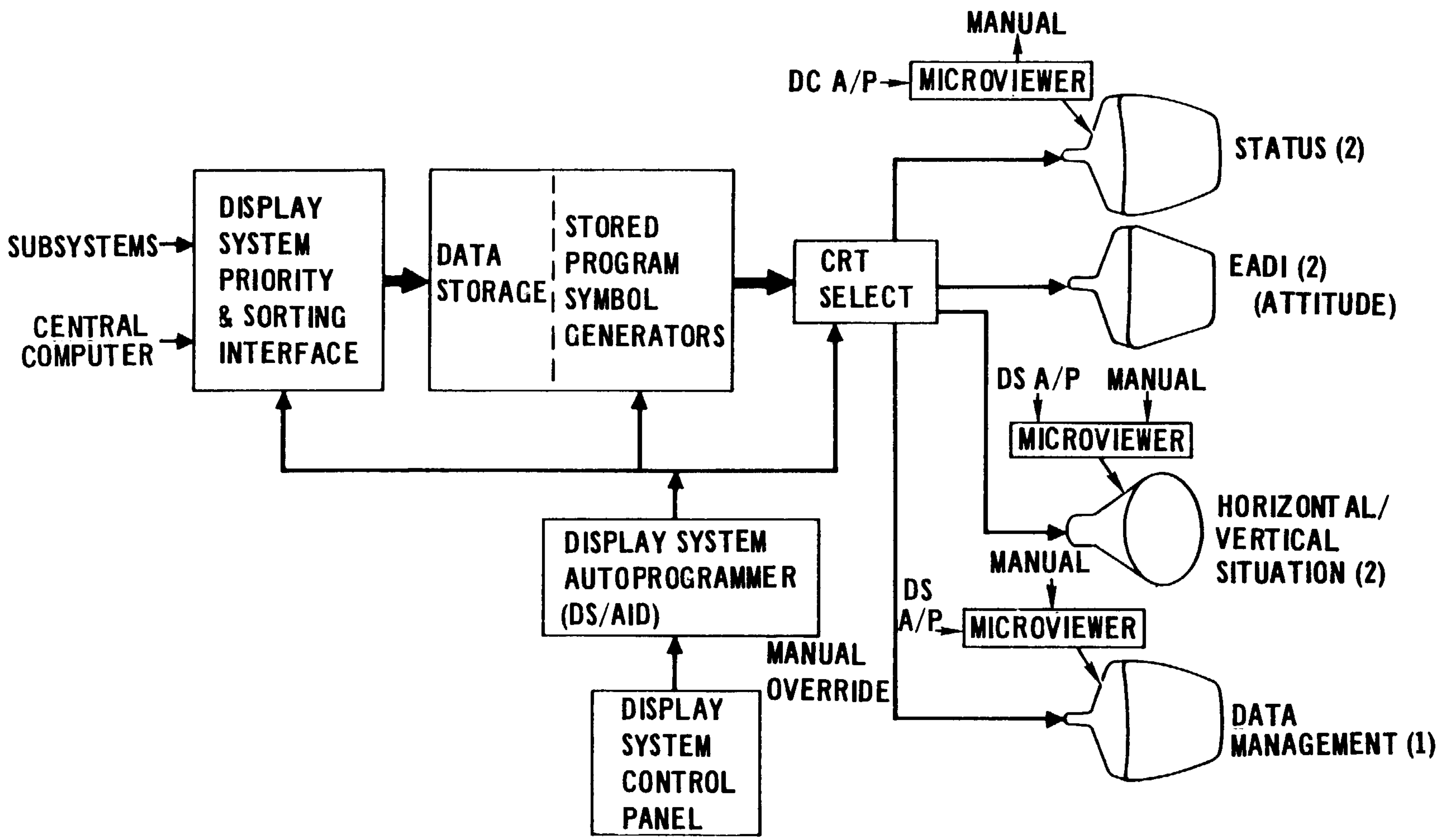
DISPLAY SYSTEM FUNCTIONAL BLOCK DIAGRAM

Four cathode ray tubes are used to provide mission operational data to each crew member. Mission information includes vehicle attitude reference, horizontal/vertical situation, operational data from onboard systems, and status of onboard systems.

The cathode ray tubes have slide or film overview capability which can accommodate large quantities of data such as diagrams of checkout procedures. These can be inserted by either autoprogrammed or manual means.

All data into the display system is through a standard interface for priority establishment and sorting to channel the proper display data storage symbology into the proper cathode ray tube. The display system data storage allows sixty cycles CRT image rewrite with a one cycle input data rate to the system.

DISPLAY SYSTEM FUNCTIONAL BLOCK DIAGRAM



ILRVS-293F

GUIDANCE, NAVIGATION AND FLIGHT CONTROL

The candidate approaches to meet key guidance, navigation, and flight control requirements are listed with the selected baseline approach and the key selection factors. The primary criteria employed in the selection process were cost, reliability and mission flexibility.

GUIDANCE NAVIGATION AND FLIGHT CONTROL

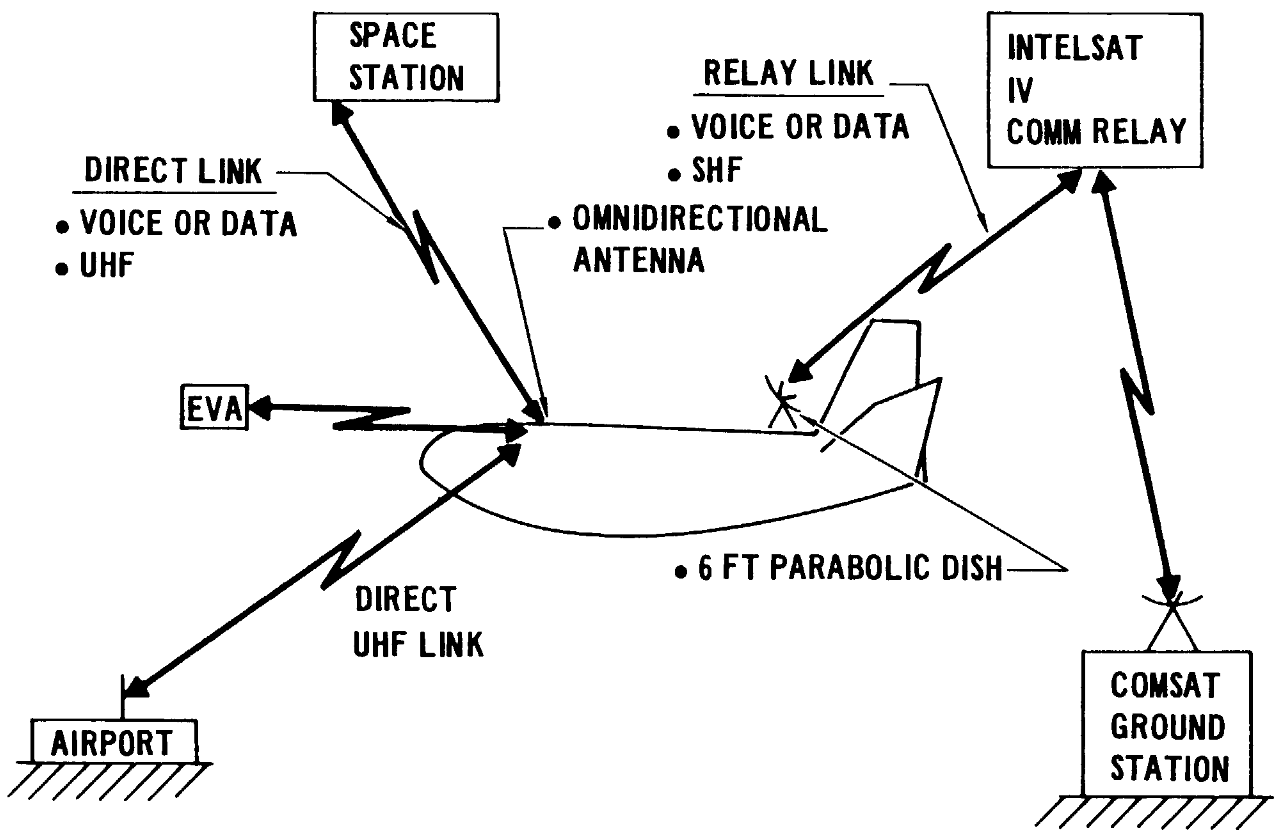
REQUIREMENTS	CANDIDATE APPROACHES & SELECTIONS	RATIONALE
SELF CONTAINED LAUNCH, ENTRY GUIDANCE	√ • STRAPDOWN INERTIAL SYSTEM • GIMBALLED INERTIAL SYSTEM √ • DEDICATED COMPUTER • CENTRALIZED COMPUTER	• RELIABILITY • SPECIALIZED USE
RENDEZVOUS	• CO-OPERATIVE (LOW POWER) RADAR √ • NON-CO-OPERATIVE RADAR • LASER RADAR • OPTICAL TRACKER	• FLEXIBILITY • ACCURACY
ON-ORBIT ALIGNMENT AND NAVIGATION	√ • AUTOMATIC STAR TRACKER • MANUAL OPTICAL √ • HORIZON SENSORS • LANDMARK TRACKER • GROUND UPDATE	• SIMPLE • MINIMIZE CREW TASKS • ACCURACY
CRUISE, FERRY AND LANDING	• INERTIAL √ • AIR DATA • RADIO COMMAND • ILS √ • ADVANCED ILS • SPN–42 • RADAR ALTIMETER √ • TACAN	• MULTI-MISSION PHASE USAGE • EXISTING EQUIPMENT • ACCURACY

√ SELECTED

TELECOMMUNICATIONS

The Intelsat IV communications relay is utilized to provide communications nearly 100 percent of the time. Use of Intelsat IV imposes a high gain (6 ft. dish) antenna and low system noise requirements on the orbiter to achieve a 3 khz voice or data bandwidth. One alternate approach is to use a dedicated UHF (225-400 mhz) satellite. At UHF there is less free space loss and omnidirectional antennas can be used on the spacecraft. However, there are possible multipath and ground interference problems. The existing TACSAT 1 relay has UHF but it will not be available.

TELECOMMUNICATIONS



ILRVS-295F

POWER SYSTEM
ORBITER

Minimum weight and reusability were the prime criteria used in selecting the electrical power source. For example, the selected orbiter power source (fuel cells and peaking/emergency batteries) weighs 1658 pounds. An equivalent source using rechargeable silver-zinc batteries would weigh approximately 19,000 pounds. Turbo alternators were disregarded as a prime electrical power source for the orbiter because of high fuel consumption and short demonstrated operating life (250 hours). Silver zinc batteries were selected for the booster power source because the short flight duration requires only 690 pounds of batteries.

Subsystem weight was the determining factor in the hydraulic power source selection. For example, a 150 horsepower electric motor weighs between 1200 and 1500 pounds, while an equivalent turbine power unit weighs approximately 80-100 pounds.

POWER SYSTEM
ORBITER

REQUIREMENTS	CANDIDATE APPROACHES	BASELINE SYSTEM
LAUNCH, ORBIT, AND ENTRY		
• ELECTRICAL	• BATTERIES • FUEL CELLS • COMBINATION FUEL CELLS/BATTERIES • APU (TURBO ALTERNATOR)	• FUEL CELLS WITH PEAKING BATTERIES FOR AVIONICS • ALTERNATORS ON APU'S FOR ROCKET ENGINE IGNITION
• HYDRAULIC	• ROCKET ENGINE DRIVEN PUMPS • ELECTRIC MOTOR/PUMP UNITS • COMBINATION • APU PUMPS	• ROCKET ENGINE DRIVEN PUMPS WITH APU PUMP BACKUP • APU PUMPS FOR AERO SURFACE OPERATION DURING ENTRY
CRUISE & LANDING		
• ELECTRICAL	• JET ENGINE DRIVEN ALTERNATORS • BATTERIES • FUEL CELLS • COMBINATION FUEL CELLS/BATTERIES • APU (TURBO ALTERNATOR)	• FUEL CELLS WITH PEAKING BATTERIES FOR AVIONICS
• HYDRAULIC	• JET ENGINE DRIVEN PUMPS • ELECTRIC MOTOR/PUMP UNITS • COMBINATION	• JET ENGINE DRIVEN PUMPS • APU DRIVEN PUMPS PRIOR TO JET ENGINE START AND AS BACK-UP

ILRVS-296F

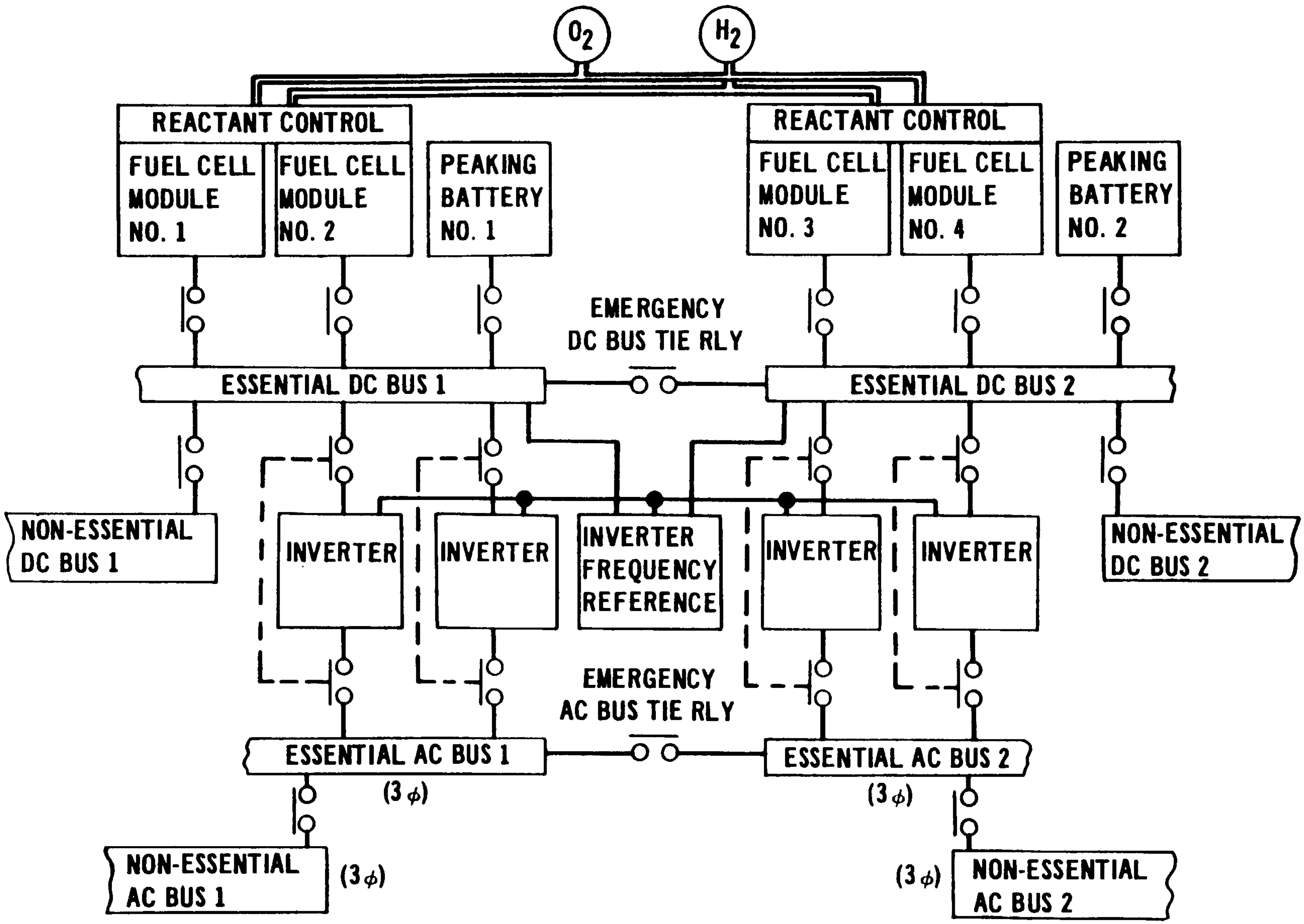
ORBITER ELECTRICAL POWER SUBSYSTEM

Mission success is one of the prime features of the electrical power subsystem. The main power source consists of four matrix type H_2-O_2 fuel cell modules for a total rated output of 10 kw. All fuel cell modules are in continuous operation for fuel economy as well as continuity of power if a fuel cell module fails. In addition to the fuel cell modules, batteries are provided for emergency deorbit and landing.

The source redundancy and busing arrangement is an adaptation of that used in commercial aircraft such as the DC-9 and DC-10. Operational flexibility is provided by employing both isolated and parallel operation of sources and buses.

The booster electrical power subsystem is essentially identical, except that silver-zinc batteries are used for the main power source instead of fuel cells because of the short flight duration.

ORBITER ELECTRICAL POWER SUBSYSTEM



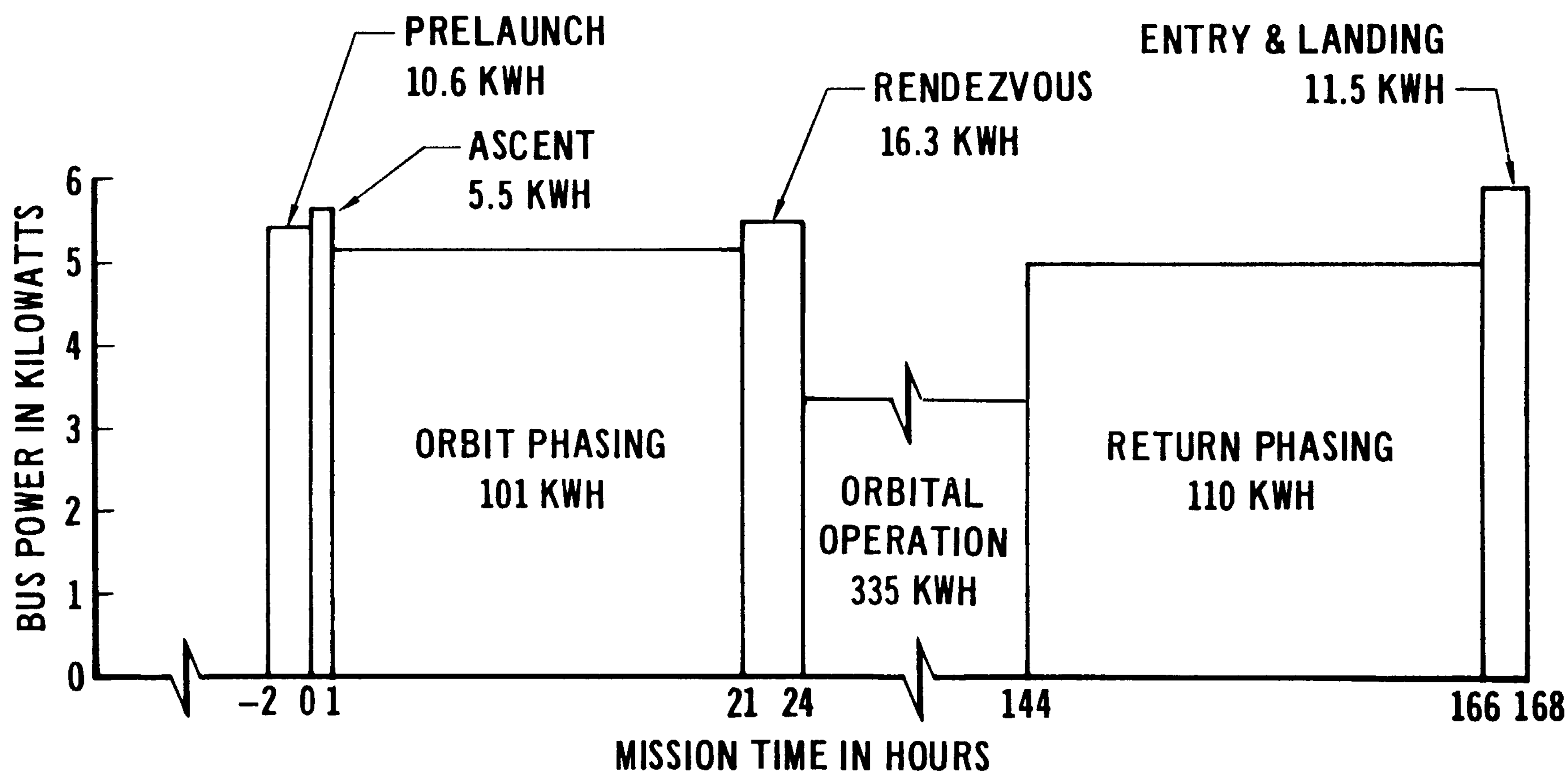
ILRVS-297F

ORBITER MAIN BUS AVERAGE POWER

The orbiter bus electrical load profile and energy requirements are shown in this chart. The variations in the main bus average power level are shown in the load profile. For simplicity, momentary or short duration peak loads are omitted, but these peaks are within the capability of the power source.

The overall average power level for the mission is 3.3 kw with 6.9 kw peaks during rendezvous and docking operations.

ORBITER MAIN BUS AVERAGE POWER
Total Mission Energy: 562.1 KWH



ILRVS-298F

BASELINE INTEGRATED AVIONICS SYSTEM

The systems selected for the orbiter are listed with those systems not required on the booster noted. The number of systems required for redundancy are also indicated.

BASELINE ORBITER INTEGRATED AVIONICS SYSTEM

COMMUNICATIONS
- SHF* (2) & UHF (2) TRANSCEIVERS
- PROCESSOR (3)
- INTERCOM & HEADSETS (2)
- OMNI ANTENNAS (4)
- *SHF DISH ANTENNA (1)

GUIDANCE AND NAVIGATION
- INERTIAL MEASUREMENT (3)
- G&N COMPUTER (3)
- *INTEGRATED OPTICAL & IR (2)
- *RENDEZVOUS RADAR (2)
- *DOCKING SENSORS (3)

LANDING & NAVIGATION AIDS
- VORTAC TRANSCEIVER (2)
- RADAR ALTIMETERS (3)
- AIR DATA SENSORS (3)
- ADVANCED ILS (3)

ELECTRICAL
- *FUEL CELL (2 OUT OF 4)
- *REACTANT SUPPLY (25%)
- BATTERIES (2)
- INVERTERS (4)

CENTRAL MANAGEMENT
- CENTRAL COMPUTER (3)
- CREW

DISPLAY & CONTROL
- MODE CONTROL PANEL (2)
- HAND CONTROLLERS (2)
- MULTI-PURPOSE CRT DISPLAYS (6)
- HEAD-UP DISPLAY (2)
- *PRINTER (1)

CHECKOUT & MONITORING
- PROPULSION PROCESSOR (3)
- INSTRUMENTATION PROCESSOR (3)
- INSTRUMENTATION SENSORS
- REMOTE MULTIPLEXERS
- FLIGHT RECORDER (2)
- *REMOTE TV CAMERAS

*DENOTES SYSTEMS NOT USED ON BOOSTER

FLIGHT CONTROL
- PROCESSOR (4)
- POWER SERVO AMPLIFIERS (4 PER FUNCTION)
- RATE GYRO (BACK-UP – 1)

ORBITER INTEGRATED AVIONICS PHYSICAL CHARACTERISTICS

Power generation and wiring are the two largest contributors to system weight. The size and weight of the booster avionics is less because of less power, wiring and slightly less avionics equipment. The peak orbiter power requirement of 5765 watts occurs during the rendezvous phase of the mission, while the peak carrier power requirement of 5042 watts occurs during the terminal landing phase.

ORBITER INTEGRATED AVIONICS PHYSICAL CHARACTERISTICS

EQUIPMENT TYPE	WEIGHT (LB)	SIZE (CU FT)	OPERATING POWER (WATTS)
GUIDANCE & NAVIGATION	720	11.8	2270
LANDING & NAVIGATION AIDS	170	3.05	460
TELECOMMUNICATIONS	325	48.85	545
CENTRAL MANAGEMENT COMPUTER	180	3.0	500
DISPLAYS, CONTROL & SEQUENCING	477	8.25	1525
FLIGHT CONTROL	75	1.3	245
CONTROL AMPLIFIERS	122	2.25	870
INSTRUMENTATION	125	2.1	260
POWER GENERATION	1658	36.0	10 KW (CAPACITY)
PWR DISTR WIRE	700	14.0	
SIGNAL DISTR WIRE	1300	20.0	
TOTAL ORBITER AVIONICS	5852	151.0	(5765) (PEAK)
(TOTAL CARRIER AVIONICS)	(4065)	(66)	(5042) (PEAK)

ILRVS-300F

RELIABILITY GOALS

The main goal is avionics operation after two failures and fail safe after the third failure. This requires on-board fault detection and switchover to redundant units. Techniques such as active failure detection, fade-in logic, modular and function redundancy are used to meet this reliability goal.

RELIABILITY GOALS

GOAL	APPROACH
• FIRST AND SECOND FAILURE – REMAIN OPERATIONAL • THIRD FAILURE – NON-CATASTROPHIC	• MODULAR AND FUNCTIONAL REDUNDANCY • FAILURE DETECTION AND SWITCHOVER
• AVOID MINIMUM PERFORMANCE BACK-UPS	• FUNCTIONAL REDUNDANCY PERMITTED ONLY WHEN MISSION PERFORMANCE IS NOT REDUCED • WHERE POSSIBLE USE EQUIPMENT ON BOARD FOR OTHER MISSION REQUIREMENTS
• MINIMIZE SYSTEM TRANSIENTS DUE TO FAILURE	• ACTIVE FAILURE DETECTION (E.G. MIDDLE SELECT) • FADE-IN LOGIC
• MISSION SUCCESS = 0.95	• HI-RELIABILITY EQUIPMENT • ON-BOARD FAULT DETECTION AND REDUNDANCY • PROGRAMMED GROUND MAINTENANCE

ILRVS-301F

TYPICAL REDUNDANCY APPLICATIONS

The degree and types of redundancy employed in the orbiter guidance and control subsystem is typical of the implementation of the reliability goals. The system uses triple redundant computers and inertial measuring units (IMR). A rate gyro is used as backup to the strapdown IMU for rate data during entry. Two rendezvous radar systems are used with backup provided from optical sensors. The optical sensors are already used to obtain navigation updates. Redundant displays and controls are provided by switching display data to operative cathode ray tubes. This large amount of redundancy provides a very high probability of successful operation.

TYPICAL REDUNDANCY APPLICATIONS
For Orbiter G & C Functions

SUBSYSTEM ELEMENT	REDUNDANCY EMPLOYED	RELIABILITY ESTIMATE
I.G.S.		
COMPUTER	DEDICATED COMPUTER (TRIPLY REDUNDANT)	.99989
I.M.U.	STRAPDOWN INERTIAL UNIT (TRIPLY REDUNDANT)	.99998
RATE GYRO PACKAGE	BACKUP R.G. PACKAGE	.99997
RENDEZVOUS SYSTEM RADAR	DUAL RADARS – OPTICAL BACKUP	.99999
TIME REFERENCE SYSTEM	DUAL – ACTIVE REDUNDANCY	.99993
STAR TRACKER HORIZON SENSOR	DUAL REDUNDANCY	.99997
DISPLAYS AND CONTROLS	100% REDUNDANT– CRT & HEADS-UP DISPLAY	.99999
TERMINAL RENDEZVOUS OPTICS	DUAL REDUNDANT OPTICAL SUBSYSTEM	.99995
TOTAL (ALLOCATION)		.99969 (.9885)

ILRVS – 302F

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MISSION INTERFACES AND CARGO ACCOMMODATIONS/HANDLING

Special Emphasis Area

ILRVS-441F

(10)

MISSION INTERFACE AND CARGO ACCOMMODATION HANDLING

The special emphasis area included the studies shown. Most of the Mission Analysis (Task 3) was completed earlier and was reported at the midterm briefing. The remainder of this analysis is included as a part of the special emphasis study.

MISSION INTERFACES AND CARGO ACCOMMODATIONS/HANDLING

- MISSION ANALYSIS
- MISSION INTERFACE DEFINITION
- CARGO HANDLING
- CREW ACCOMMODATION
- EXTENDED MISSION DURATION

ILRVS-4

MISSION PROFILE

A typical in-plane mission flight profile of the ascent and orbiter reentry phases are illustrated in this chart. The view is from the westerly direction, normal to an orbital plane that is inclined 55 degrees to the equitorial plane.

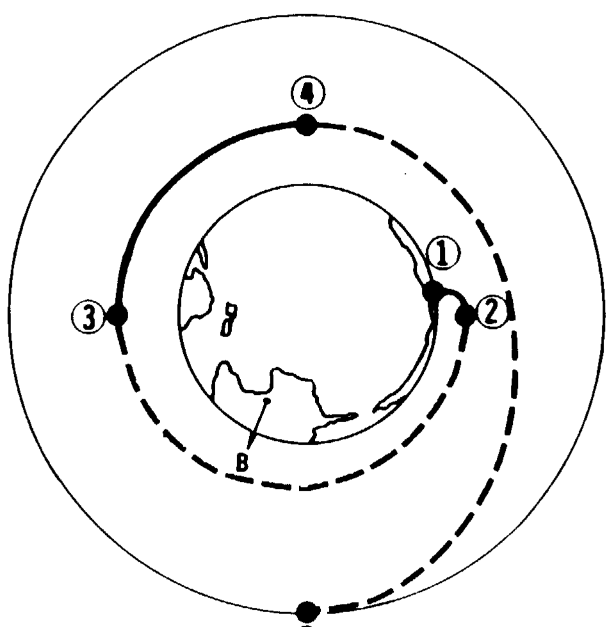
Key ascent phase events are identified as 1) liftoff, 2) second stage insertion at 45 NM perigee, 3) a Hohmann transfer to 100 NM followed by a circularization impulsive, 4) an impulsive burn after the desired phasing angle has been completed, and 5) circularization at 270 NM. The powered flight trajectory has been shaped to minimize ascent losses consistent with overall mission requirements, including safe carrier recovery.

Orbiter reentry is characterized by 1) a retrograde burn at 270 NM, 2) reentry at 400,000 feet, 3) acquisition of the desired equilibrium glide path, and 4) touchdown at the launch site. Reentry shaping has been performed within specified heating and loading constraints.

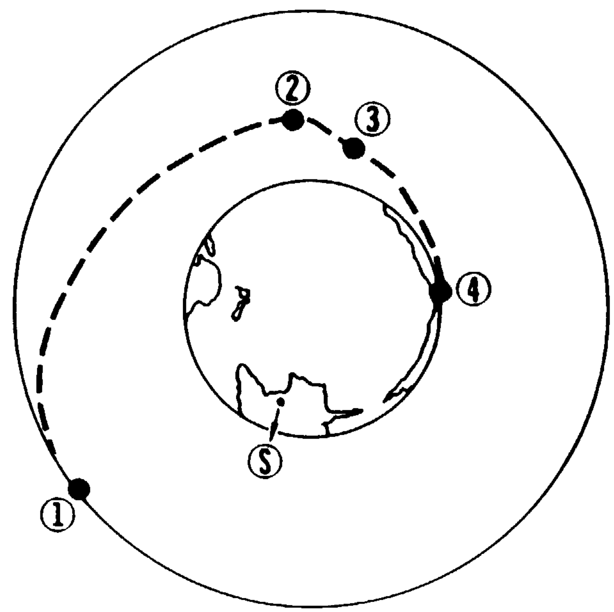
MISSION PROFILE

	t (MIN)	V (FPS)	h (NM)	R (NM)
① LIFT-OFF	T_0	0	0	0
② INSERTION	$T_0 + 7.2$	25885	45	655
③ CIRCULARIZATION	$T_0 + 50.7$	25583	100	10840
④ COMPLETE PHASING	T_P	25881	100	0
⑤ CIRCULARIZATION	$T_P + 45.6$	24990	270	10122

	t (MIN)	V (FPS)	h (NM)	R (NM)	γ (DEG)
① RETROGRADE	$T_E - 31.5$	24510	270	−7800	0
② RE-ENTRY	T_E	25990	65.8	0	−1.5
③ EQUILIBRIUM GLIDE	$T_E + 5.2$	25780	40.2	1314	−0.1
④ LANDING	$T_E + 29.3$	289	0	5568	0



Ascent



Reentry

ILRVS-404F

POST INJECTION ΔV REQUIREMENTS

In-orbit velocity increments for the baseline mission boost injection, made at perigee of a 45 x 100 NM orbit having an inclination of 55 degrees, are as follows:

Circularization - A 100 NM phasing orbit, requiring a ΔV of 100 fps, was selected to attain a desirable catch-up rate with minimum drag decay.

Catch-up - A loss of 40 fps results from drag decay during maximum phasing of 24 hours.

Orbit Transfer and Plane Change - With no plane change, the transfer maneuver to 270 NM requires approximately 540 fps and 180 fps to the 150 NM orbit. For a launch window of +60 seconds, the plane change required is about 0.2 degrees for an additional ΔV of 10 fps.

Terminal Rendezvous - Using a theoretical point value, 130 fps is required to bring the shuttle within a few hundred feet of the station. Gemini flight experience indicates a ΔV of 1-1/2 to 4 times the theoretical value was used for rendezvous. Therefore, a total of 200 fps was included in the table.

Return Phasing - Crossrange capability of approximately 500 NM is required for once-a-day return from a 55 degree orbit. Without sufficient atmospheric maneuver capabilities, orbit phasing must be used to insure that the track passes over the landing site. About 285 fps for a 270 NM orbit is required to shift the orbit track 12 degrees, which is the worse case.

Retrograde - The ΔV needed is 425 fps for a 270 NM orbit and 250 fps for a 150 NM orbit.

Recommendation - Ten percent ΔV contingency appears adequate which brings the recommended total to 1490-1805 fps for the 270 NM orbit. Deployment to 150 NM can be performed with less, thus increasing the payload at the rate of approximately 21 pounds per fps.

POST INJECTION ΔV REQUIREMENTS

FUNCTION	STAGE/SATELLITE DEPLOYMENT (150 N.M.)	SPACE STATION RENDEZVOUS (270 N.M.)	NASA SPECIFIED ΔV
• CIRCULARIZATION	100	100	100
• CATCH-UP	40	40	–
• ORBIT TRANSFER	180	540	558
• PLANE CHANGE	10	10	–
• LAUNCH DISPERSION	0	0	200
• RENDEZVOUS	0	200	142
• STATION KEEPING	40	40	–
• ENTRY PHASING	0–285*	0–285*	–
• RETROGRADE	250	425	500
SUBTOTAL	620–905	1355–1640	1500
• CONTINGENCY	60–90	135–165	500
TOTAL ORBIT ΔV	680–995	1490–1805	2000

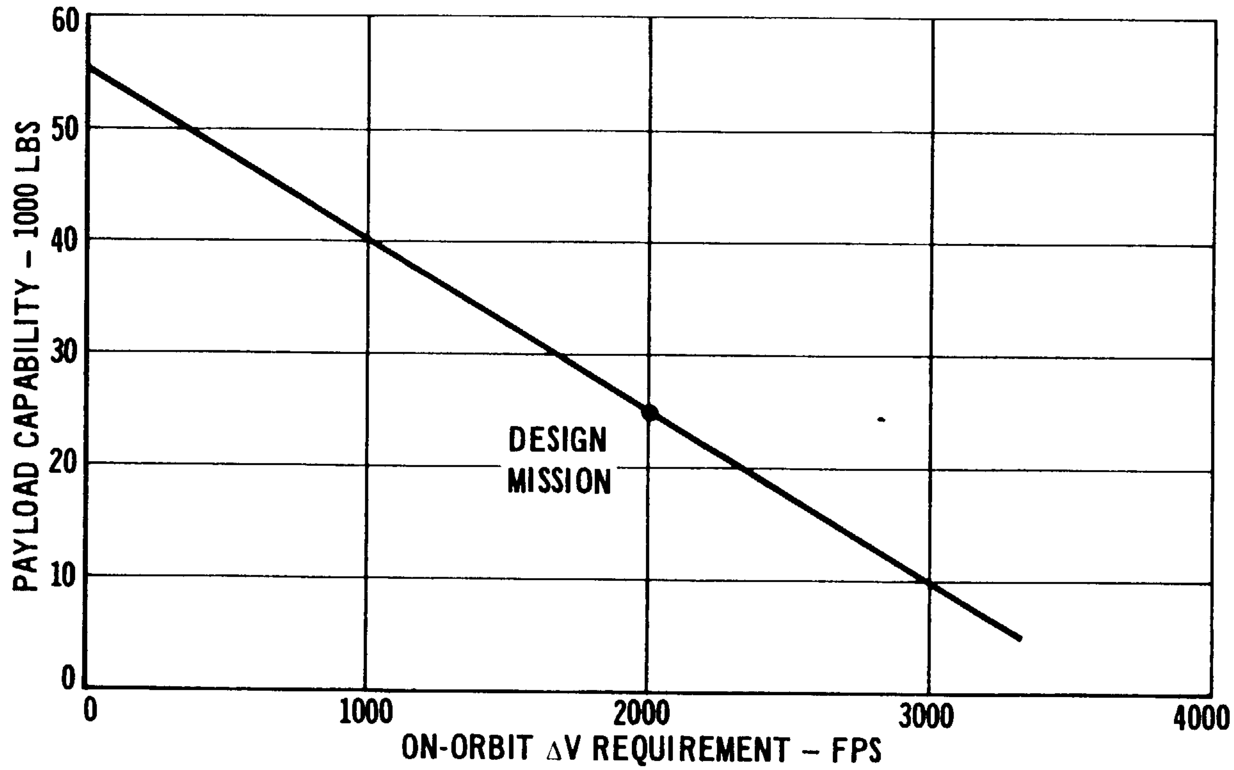
*FUNCTION OF CROSS-RANGE CAPABILITY

ILRVS–469F

PAYLOAD SENSITIVITY TO ON-ORBIT ΔV REQUIREMENTS

The sensitivity of payload weight ot variation in on-orbit ΔV required is shown on the facing page. The exchange ratio is approximately 15 lb (cargo) per fps change in ΔV. Thus, if the ΔV requirements were reduced by 100 fps from the nominal 2000 fps, 1500 lb of cargo could be added. Similarly, an increase of 100 fps in ΔV required would result in a decrease of approximately 1500 lb of cargo.

PAYLOAD SENSITIVITY TO ON-ORBIT ΔV REQUIREMENTS

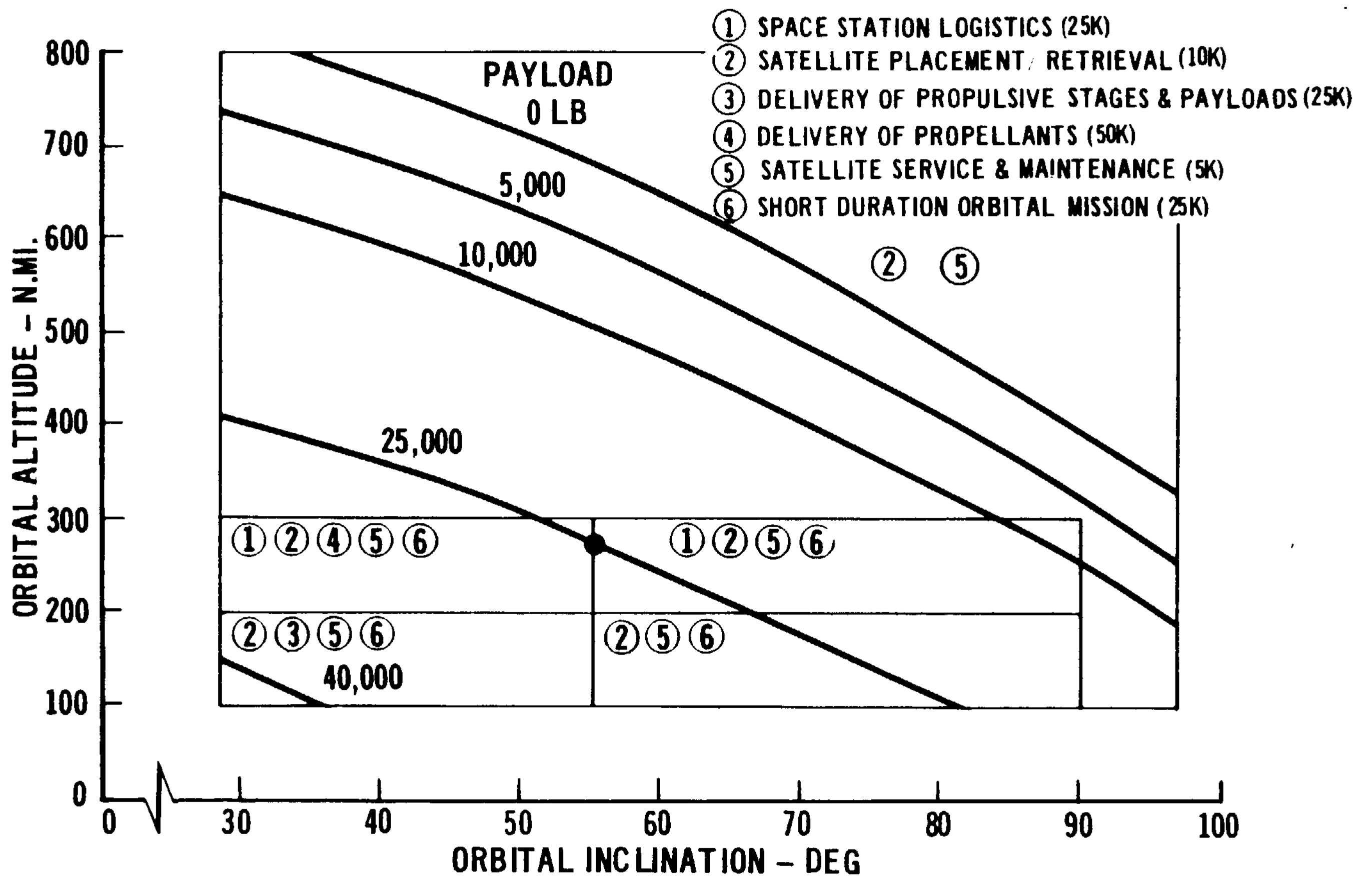


ILRVS-232F

MISSION PERFORMANCE CAPABILITY

The flexibility of a space shuttle system to perform missions other than the design reference mission (55-degrees inclination at 270-NM altitude) can be determined by trading payload weight for maneuvering fuel. The figure shows the capability of the reference system in terms of constant payload as a function of altitude and inclination. The altitude and inclination grid is divided into rectangular areas, identified by numbers indicating those missions that may be performed in that given area. The minimum payload required for each mission is indicated in parenthesis after each alternate mission definition. At altitudes less than 270 NM and inclinations between 28.5 and 55 degrees, payloads in excess of the 25,000-pound design payload are possible; while at higher altitudes and inclinations, the payload capability decreases. For example, at 90-degrees inclination and 320-NM altitude, in the region where the most probable missions would be Satellite Placement/Retrieval or Satellite Service and Maintenance, the payload has decreased to 5,000 pounds due to the amount of fuel required to satisfy the orbited conditions.

MISSION PERFORMANCE CAPABILITY



MAJOR INTERFACE AREAS

In the investigation and definition of mission interfaces, seven major mission-interface related areas were encountered. These are outlined on the adjacent page and are addressed more extensively on the following pages.

MAJOR INTERFACE AREAS

- ON-ORBIT PAYLOAD UNLOADING
- ON-ORBIT PAYLOAD TRANSFER (TWO WAYS)
- ON-ORBIT PAYLOAD LOADING
- CREW-ACCESS TUNNEL

ILRVS 237F

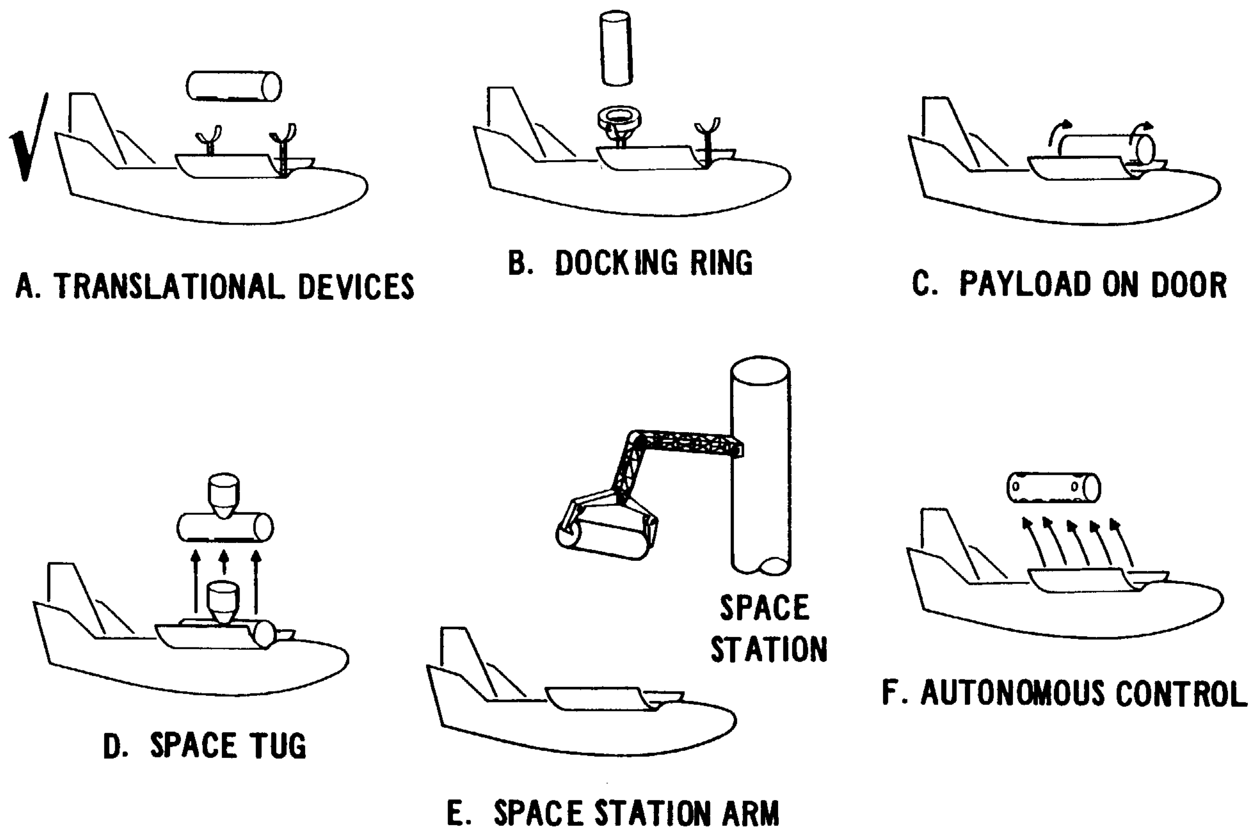
METHODS OF PAYLOAD UNLOADING

The various methods and techniques considered for extracting the cargo module from the cargo bay of the Orbiter are illustrated in the accompanying chart. These include:

- o Translational Devices
- o Swing-out Docking Ring
- o Payload swings out with door opening
- o Space Tug pulls payload out
- o Space Station "cherry picker" pulls payload out
- o Payload removes self through use of propulsive devices

On the basis of simplicity, use with alternate missions, and minimum dynamic force considerations, the use of a two-way translational and holding device for removing the payload from the orbiter is selected as the preferred approach.

METHODS OF PAYLOAD UNLOADING



ILRVS-238F

METHODS OF ON-ORBIT PAYLOAD TRANSFER

Four possible methods for transferring the payload from the Orbiter to the Space Station are illustrated on the opposite page. These include:

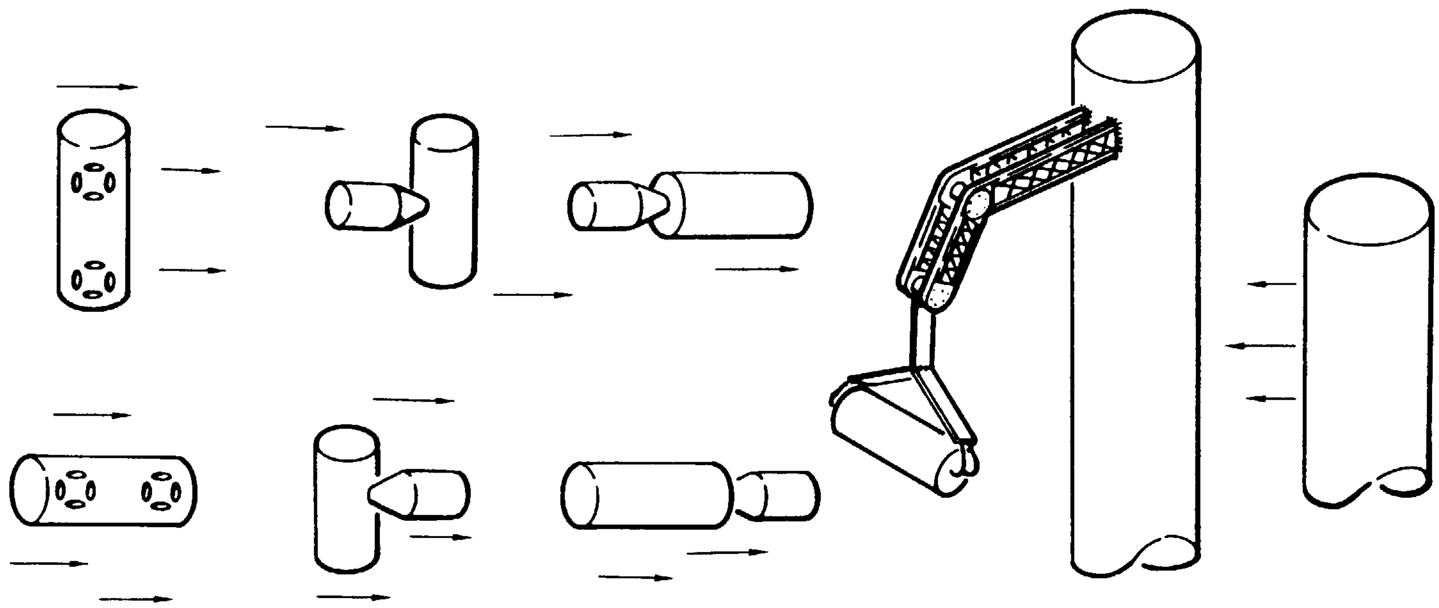
1. autonomous control (onboard RCS)
2. use of space tug (either pushing or pulling)
3. use of space station "arm"
4. Space Station comes to payload

An assessment was made as to the advantages and disadvantages of each of the transfer methods and the preliminary conclusion is that either method 1 or 2 could be selected as the preferred mode. Method 2, use of the Space Tug (pushing), was finally selected as the preferred system on the grounds that it did not compromise payload capability to the extent that the autonomous control method did.

This particular operation and the methods of implementing it warrants more in-depth analysis.

Note: A ground rule established in investigating all payload unloading and transferring techniques was that no docking was to take place between the Orbiter and Space Station or between the payload (while still physically attached to the Orbiter) and Space Station. This ground rule is entirely reasonable when one considers the size of the proposed Space Station as compared with the size of the Orbiter plus payload, particularly during the early build-up phases of the Station. In later stages of Space Station build-up a portion of the station volume may be rotating. Again, the dynamics implications of large added masses suggest the "no-dock" mode.

METHODS OF ON-ORBIT PAYLOAD TRANSFER



A. AUTONOMOUS CONTROL

✓ B. USE OF SPACE TUG

C. SPACE STATION ARM

D. SPACE STATION COMES TO PAYLOAD

ILRVS-239F

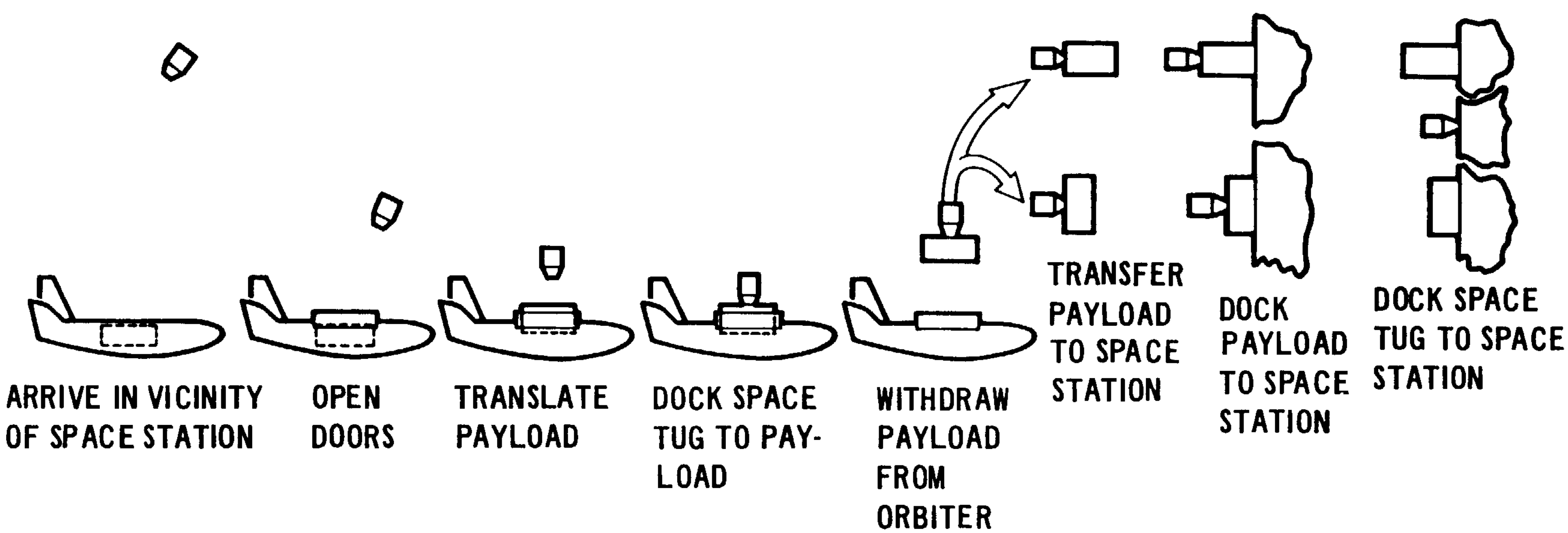
PAYLOAD TRANSFER SEQUENCE - USE OF SPACE TUG TECHNIQUE

The sequence of illustrations on the facing page shows pictorically the event-to-event procedure for unloading and transferring the payload from the Orbiter to the Space Station. This technique was selected as the preferred approach with some reservations as stated in the preceding paragraphs.

The figure shows how the translation of the payload beyond the limits of the open doors is accomplished before the Space Tug docks with the payload, thus demonstrating the interplay between major systems in the operational sequence of events.

PAYLOAD TRANSFER SEQUENCE

Space Tug



CREW-ACCESS TUNNEL ASSESSMENT

The advantages and disadvantages of incorporating a crew-access tunnel into the design of the baseline vehicle were weighed against each other. The crew-access tunnel would allow the crew to have in-orbit access to the cargo bay and the payload canister. It would also require pressurization.

On the basis of the above-noted assessment, which is detailed on the facing page, it was decided to provide a crew-access tunnel aboard the Orbiter.

CREW -ACCESS TUNNEL ASSESSMENT

ADVANTAGES

- CREW HAS ACCESS TO CARGO FOR ON-ORBIT OPERATIONS AND ALTERNATE MISSION CAPABILITY
- PROVIDES CREW-TRANSFER-TO-SPACE STATION CAPABILITY VIA CARGO MODULE
- POSSIBLE ALTERNATE ESCAPE ROUTE DURING ABORT SITUATIONS

DISADVANTAGES

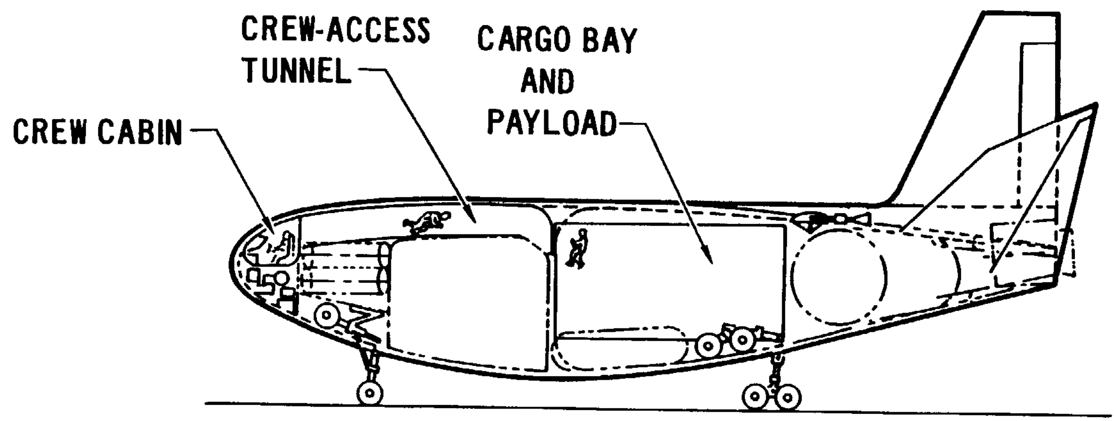
- REQUIRES ADDITIONAL PRESSURIZATION AND POWER
- MAY INTERFERE WITH PROPELLANT TANK PLACEMENT
- MAY REQUIRE PLACEMENT OUTSIDE OF ORBITER MOLDLINE
- USES VOLUME OTHERWISE AVAILABLE FOR ORBITER SYSTEMS

ILRVS-242F

CREW-ACCESS TUNNEL PLACEMENT

The figure on the opposite page shows the placement of the crew-access tunnel aboard the HL-10 Orbiter. The tunnel will be pressurized with connections at either end to the crew cabin and the cargo bay. At the cargo bay end, the capability is provided for the crew to transfer into the payload canister while in orbit.

CREW-ACCESS TUNNEL PLACEMENT



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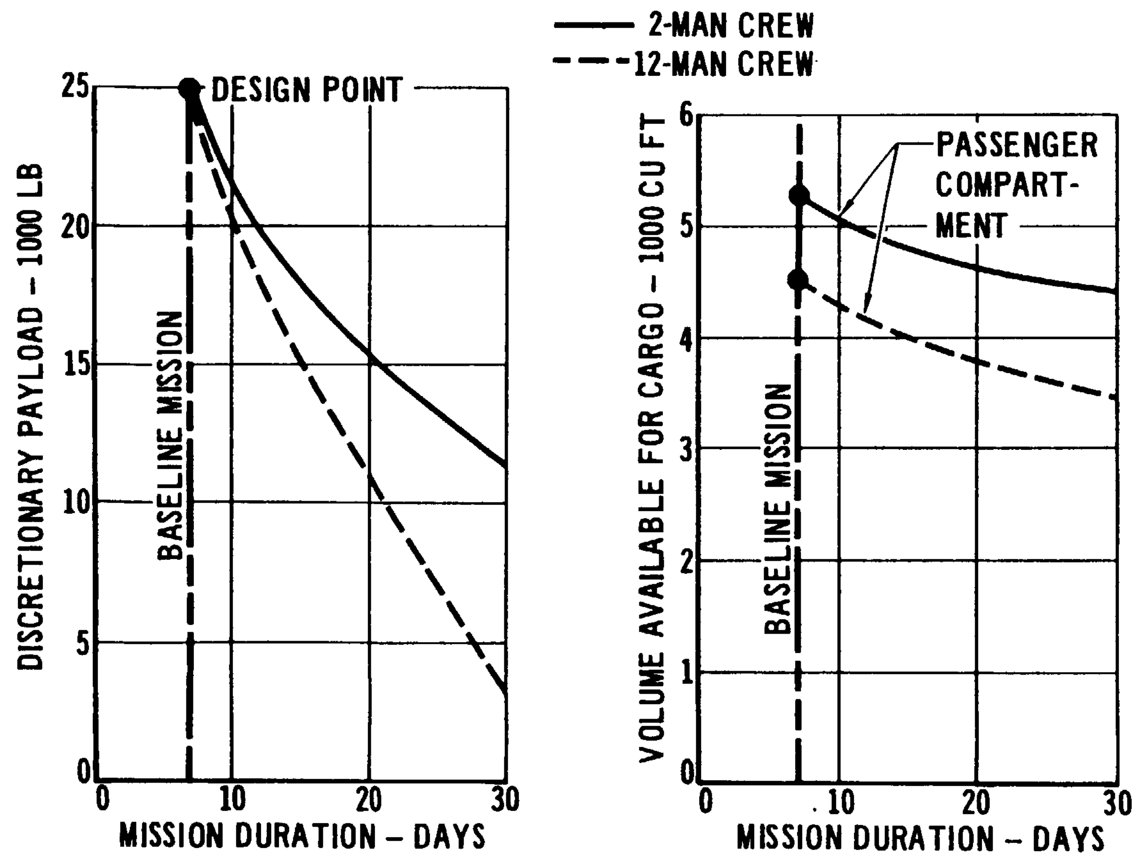
EFFECT OF MISSION DURATION ON PAYLOAD

The overall weight and volume effects of increasing the length of the mission from 7 to 30 days are seen in the charts on the facing page. Here, the increases in weight and volume of the Orbiter subsystems are charged against the baseline 25,000-pound, 5,300-cu. ft. payload. Also shown are the payload differences resulting from an assumption of a 2-man crew versus a 12-man crew, and the combined effects of both variables.

It is seen that the available Orbiter payload erodes very quickly with an increase in mission duration. In particular, a decrease from 25,000 lbs. to 11,400 lbs. results for the 2-man crew when going from 7 to 30 days. Similarly, the 12-man-crew mission sees a payload decrease from 25,000 lbs. to 3,400 lbs. over the same range of mission length.

In the case of the cargo volume, the decrease in available volume is not nearly as great as was the weight, with the 12-man-crew mission showing somewhat greater effects. However, in neither case does the payload volume diminish sufficiently to warrant curtailment of the mission duration.

EFFECT OF MISSION DURATION ON PAYLOAD



ILRVS-362F

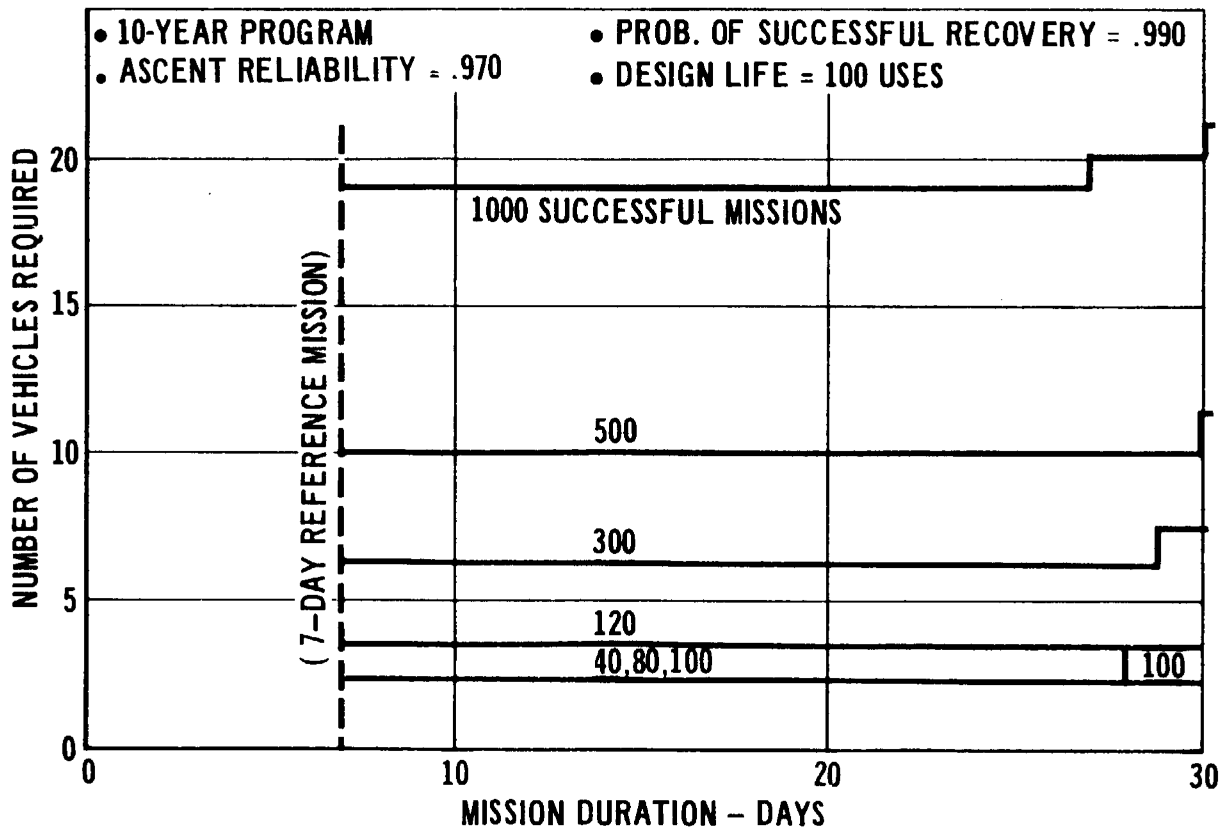
EFFECT OF MISSION DURATION ON SPACECRAFT INVENTORY

The plot shown on the facing page shows the effects of increasing mission duration on basic Orbiter spacecraft inventory. Since the Carrier continues to have a "mission" duration of less than 1 day, no change in its inventory occurs.

Changes in the Orbiter inventory occur with mission duration only when it is assumed that all missions have the particular mission length in question. Mixes of different mission durations were not investigated. However, it is seen that for missions of a duration of 27 days or less, no increase in Orbiter inventory occurs even for launch rates of 100 launches per year (1000 successful missions).

Note: The spacecraft inventories are sensitive to the inputs of design life, mission reliability, probabilities of successful recovery, etc. Sensitivity to these inputs are shown elsewhere in this report.

EFFECT OF MISSION DURATION ON SPACECRAFT INVENTORY (ORBITER ONLY)



ILRVS – 244F

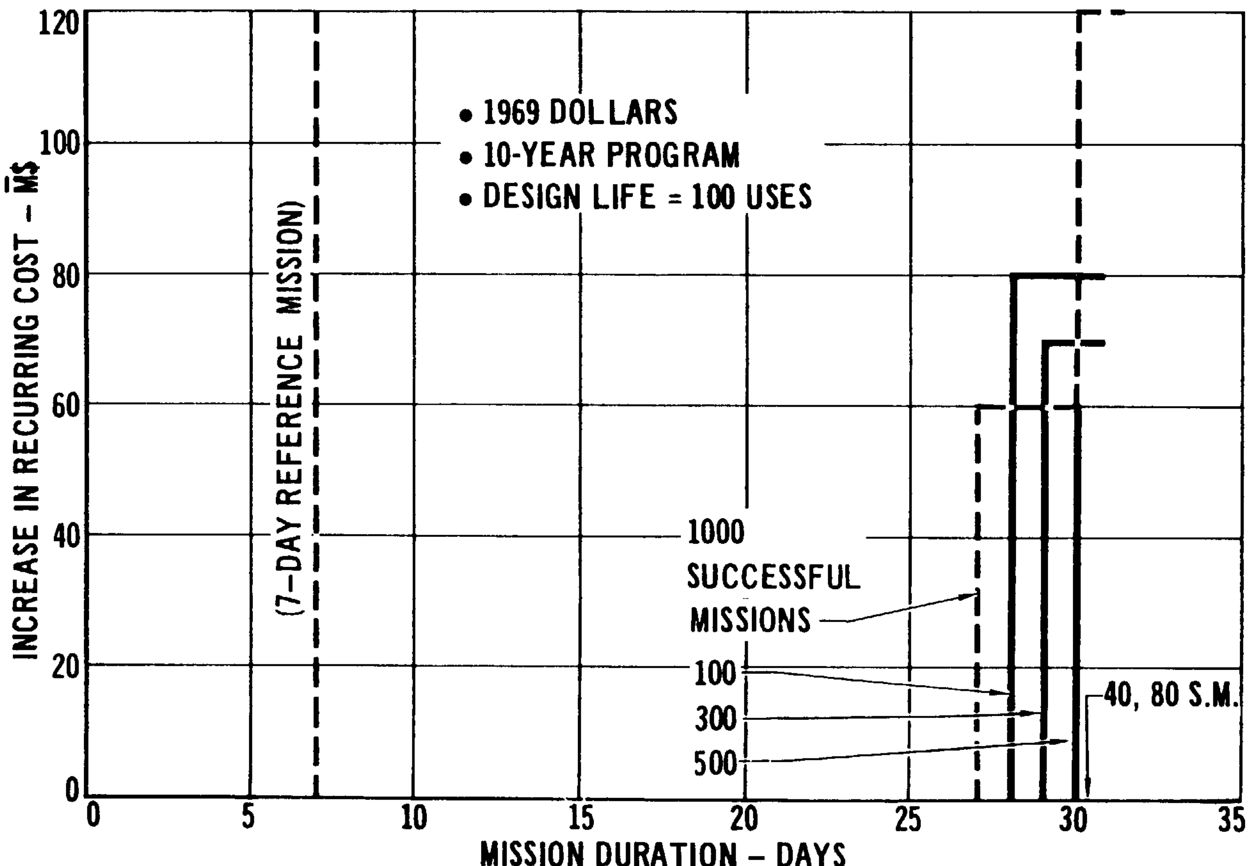
EFFECT OF MISSION DURATION ON RECURRING COST

It was shown in the previous set of charts, that the effect of increasing the space shuttle mission from 7 to 30 days is realized by an increase in the number of vehicles required to complete the program. This result is predicated on the assumption that all the missions are increased by the number of days in question. The exact increase in the number of vehicles was seen to vary with the launch rate.

The chart on the opposite page shows the recurring cost differences resulting from the increased vehicle requirement. Since an increase in mission duration can be accomplished without a corresponding increase in the Orbiter's subsystem hardware (save for a few propellant, gas, and reactant tanks), no increase occurs in the spacecraft basic unit cost. Thus all cost increases shown are the result of increased inventories, which in turn, are the result of increased mission length.

The increase in recurring costs are all seen to be less than $120 million for any of the programs considered. No cost increases occur for the 40, 80, or 120 successful mission programs, and none occur for missionsless than 27 days regardless of launch rate (for the range investigated).

EFFECT OF MISSION DURATION ON RECURRING COSTS

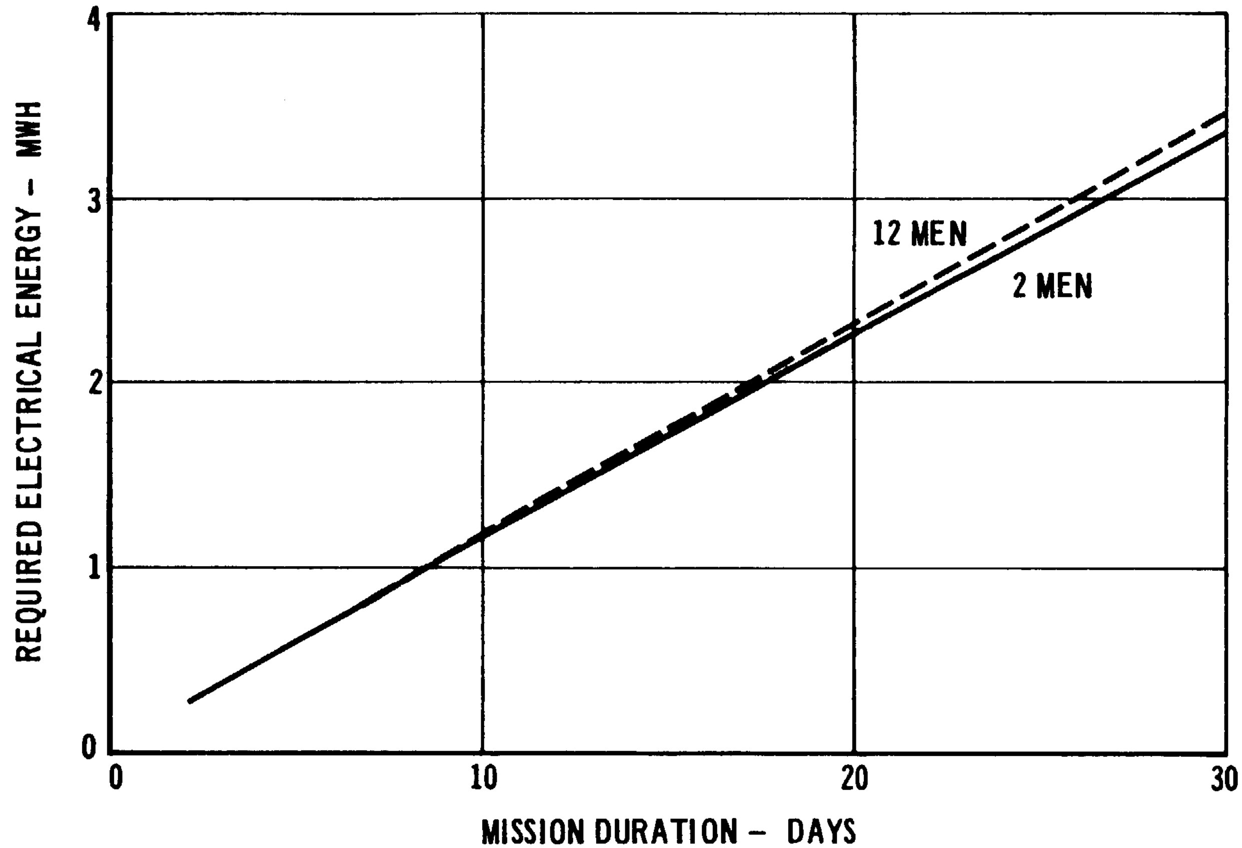


ILRVS-392F

EFFECT OF MISSION DURATION ON ORBITER ELECTRICAL POWER

The plots shown on the opposite page illustrate the increase in electrical energy requirements with increase in mission duration. The linear relationship stems from the assumption of a constant 110 kilowatt-hour basic requirement for each day of the operational (on-orbit) mission. Imposed on top of these requirements is the assumption of a 200 kilowatt-hr/day/man requirement for the operation of experiments (one experiment per crewman, average power of 100 watts, run for 4 hours per day). This accounts for the difference in energy requirements between the 2-man and the 12-man missions.

EFFECT OF MISSION DURATION ON ORBITER ELECTRICAL POWER REQUIREMENTS



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GROUND TURNAROUND OPERATIONS

Special Emphasis Area

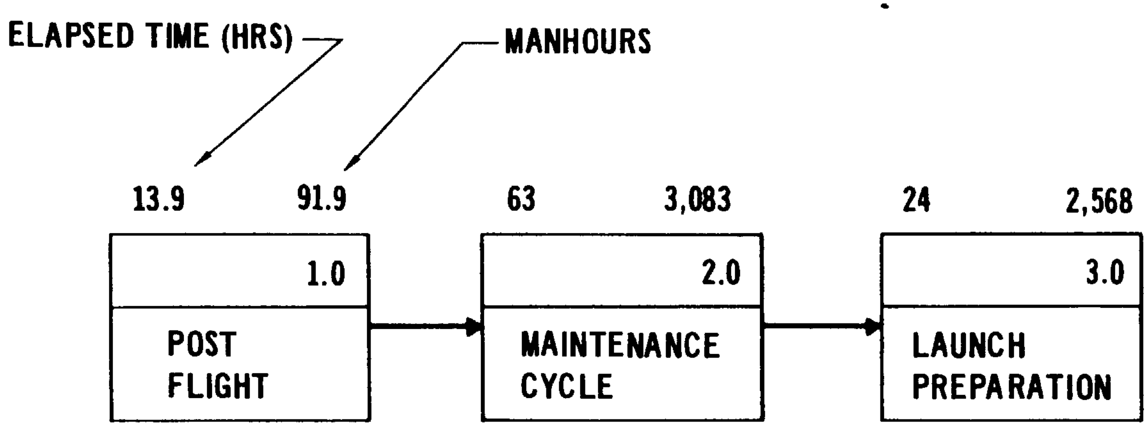
ILRVS-442F

11

TURN-AROUND CYCLE FUNCTIONAL FLOW CHART

The turn-around cycle is divided into three top level functions. The numbers above each block represents hours to completion of task. The number on the left side of the block is elapsed hours while the number on the right is manhours. Each top level function block has been expanded to levels identifing tasks of the system and subsystem component level.

TURNAROUND CYCLE
FUNCTIONAL FLOW CHART



ILRVS-387F

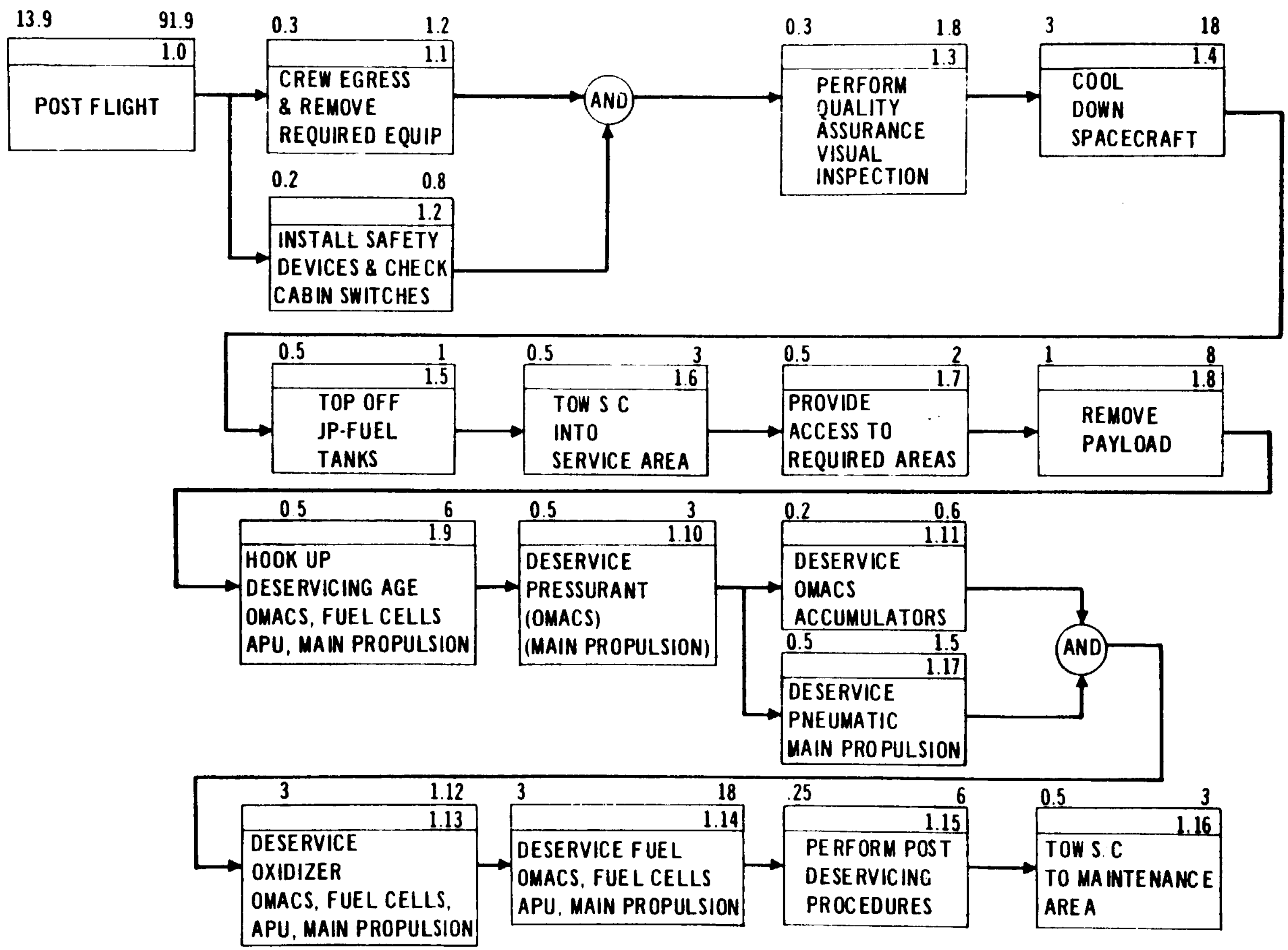
POST FLIGHT

This chart identifies necessary tasks required during post flight.

Elapsed time and manhours are identified with each task and summarized above the post flight reference block. When tasks are performed in parallel, the longest elapsed time is used for the summary, however, the manhours are added (e.g., Tasks 1.1 and 1.2 times input would be .3 elapsed time and 2.0 manhours).

These tasks have been analyzed in detail and timelined by studying airline maintenance plans, maintainability, and programs developed for DC-8, DC-9 and DC-10. Inspections and maintenance approaches used in the X-15, F-4 and C-5A were also evaluated.

POST FLIGHT



IL RVS-388 F

POST LANDING

A description of the ground turnaround tasks is shown on the left of the chart.

After landing, the vehicle taxis to a parking area adjacent to the service area for crew egress, cool down, immediate visual inspection, and JP fuel tanks top off after cool down.

SERVICE AREA

At the service area the payload is removed, access provided to required areas, and the orbital maneuvering attitude control system (OMACS) main propulsion system, auxiliary power unit (APU) and fuel cells are deserviced. After completing these tasks in the service area, the vehicle will be moved to the maintenance area.

MAINTENANCE AREA

The maintenance area is where the bulk of the subsystem scheduled and unscheduled maintenance and functional checks take place. For example, structure, heat protection, and landing equipment is inspected and repaired or replaced as required. Upon completion of these maintenance tasks, the vehicle will be moved to the air breathing engine run up area.

RUN UP AREA

Here the air breathing engines are run up and certified and subsystem operational checks performed. At this time, subsystems such as communications, radar, ILS/VOR are checked for noise, power output, or interference under operating conditions. JP fuel tanks are topped off in this area after run up.

POST MAINTENANCE AREA

At a predetermined time, the payload is transported to the post maintenance area and positioned. When the air breathing engines and subsystems are certified, the vehicle is moved into the post maintenance area where the air breathing engine maintenance tasks (flush, dry bearings and apply dry lubrication) are performed, the payload is installed and the spacecraft is positioned on the erection dolly.

STAGING AREA

The staging area is an enclosure that provides protection for a spacecraft vehicle pending its delivery to the launch pad.

PAD DELIVERY

Upon notification for launch preparations, each stage will be moved to the launch pad on its erection dolly and prelaunch activities commence.

(1) POST LIGHT
PARKING AREA
CREW EGRESS
COOL & DECONTAMINATE
QC INSPECTION
TOP JP FUEL

(2) SERVICE AREA
REMOVE PAYLOAD
PROVIDE ACCESS
DESERVICE ACS
DESERVICE BOOST SYSTEM
DESERVICE FUEL CELL
DESERVICE APU
TOW TO MAINT AREA

MAINT CYCLE
(3) • MAINT AREA
POSITION AGE
PROVIDE ACCESS
SCHEDULE MAINT
UNSCHEDULE MAINT
FUNCTIONAL CHECKS

(4) • AIR ENG RUN UP
RUN UP ENG
OPERATIONALLY CHECK
SUBSYSTEMS
TOP JP FUEL

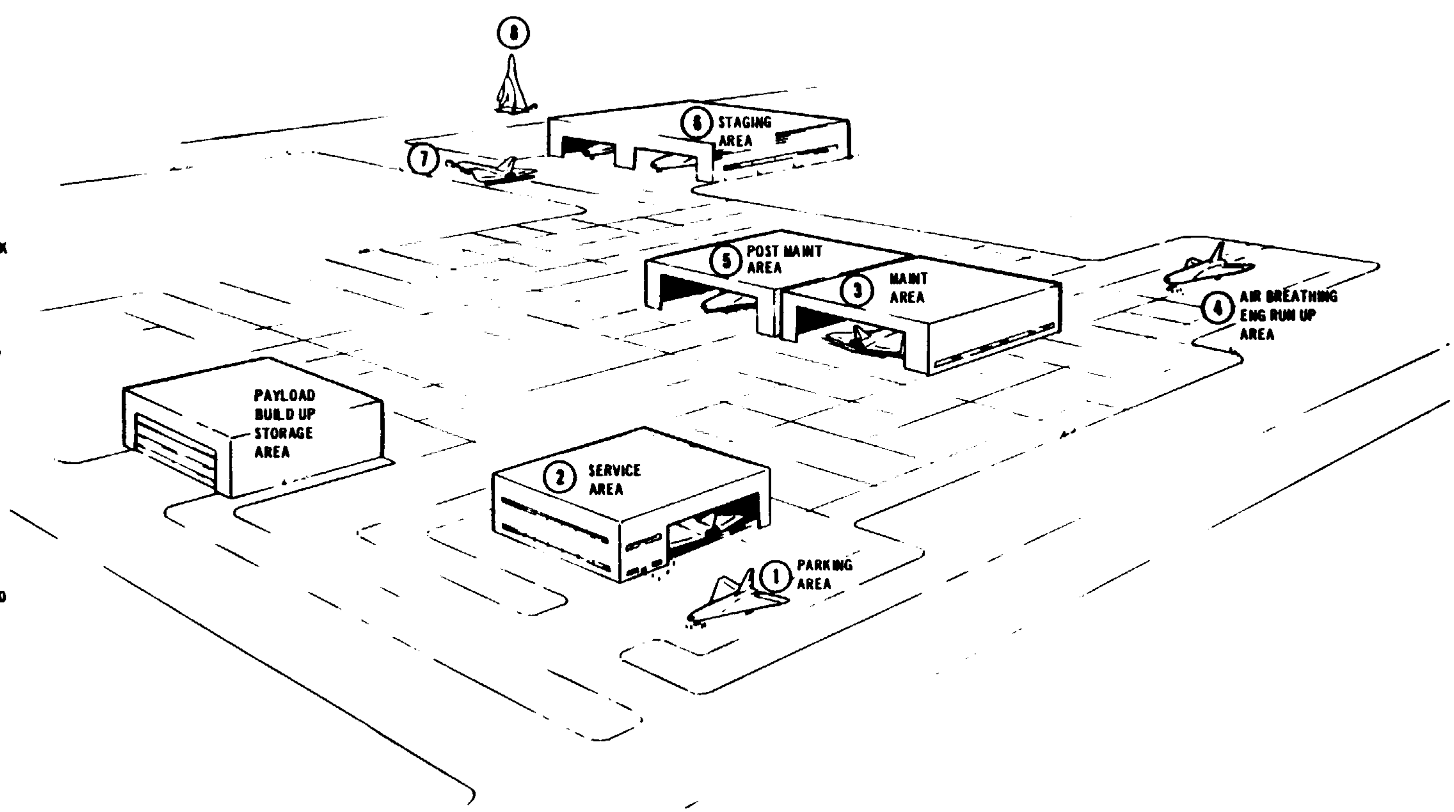
(5) • POST MAINT AREA
AIR BREAT ENG TASKS
INSTALL S/C ON DOLLY
INSTALL PAYLOAD
SERVICE

(6) STAGING AREA
PROTECTIVE AREA
FOR VEHICLES
WAITING TO BE
DELIVERED TO PAD

(7) LAUNCH PREP
1ST STAGE TOWED
TO PAD
2ND STAGE COMPLETED
MAINT IN STAGING
AREA WILL FOLLOW
WHEN 1ST STAGE IS
ERECTED

(8) LAUNCH PREP
BOTH STAGES
ERECTED AND
PROCEEDING WITH
COUNTDOWN

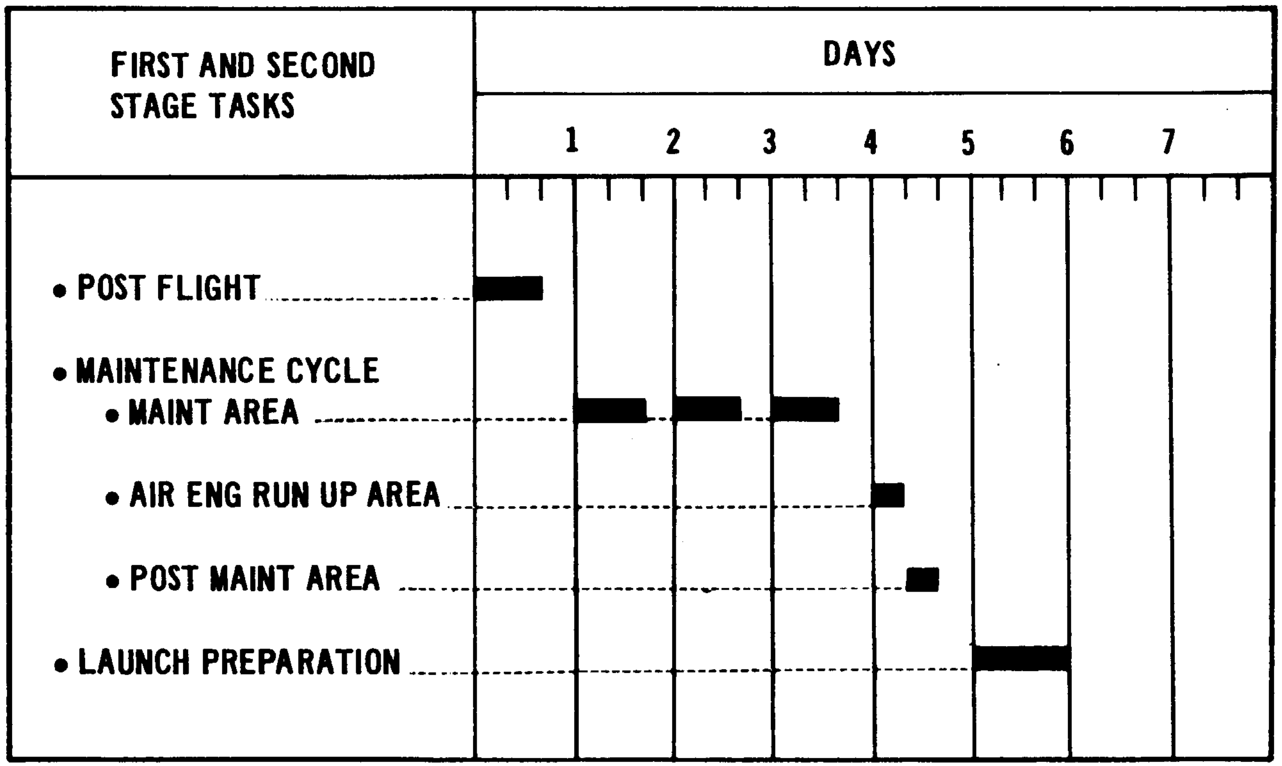
GROUND TURN-AROUND



MINIMUM TURN-AROUND SUMMARY

As a result of the detailed functional flow analyses, utilizing two 8-hours shifts per day during the post flight and maintenance cycle, the minimum turn-around cycle can be accomplished within six days. This rapid turn-around cycle requires 360 men. At the beginning of the program, with a low traffic rate, the turn-around cycle can be accomplished within 27 days employing 90 men working one 8-hour shift per day.

MINIMUM TURNAROUND SUMMARY



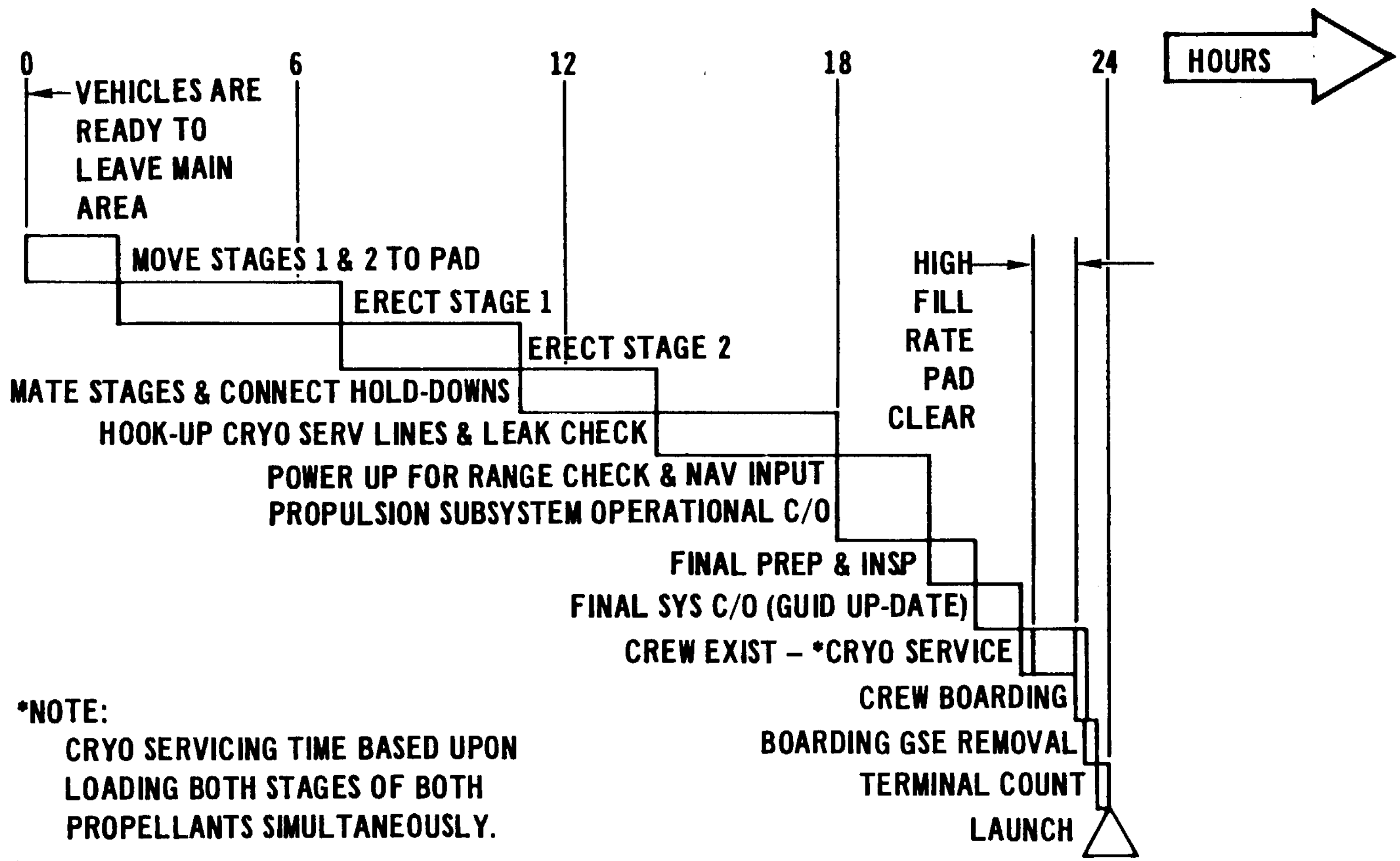
STAFFING 720 MEN
TOTAL MANHOURS PER LAUNCH 17,102

ILRVS-389F

LAUNCH OPERATIONS SCHEDULE

The Operations Schedule presents the sequence of events within the final twenty-four hour period prior to launch. These activities are based on the on-pad build-up technique. For pre-pad build-up the basic schedule difference would be the transport, tie down and service connection times.

LAUNCH OPERATIONS SCHEDULE

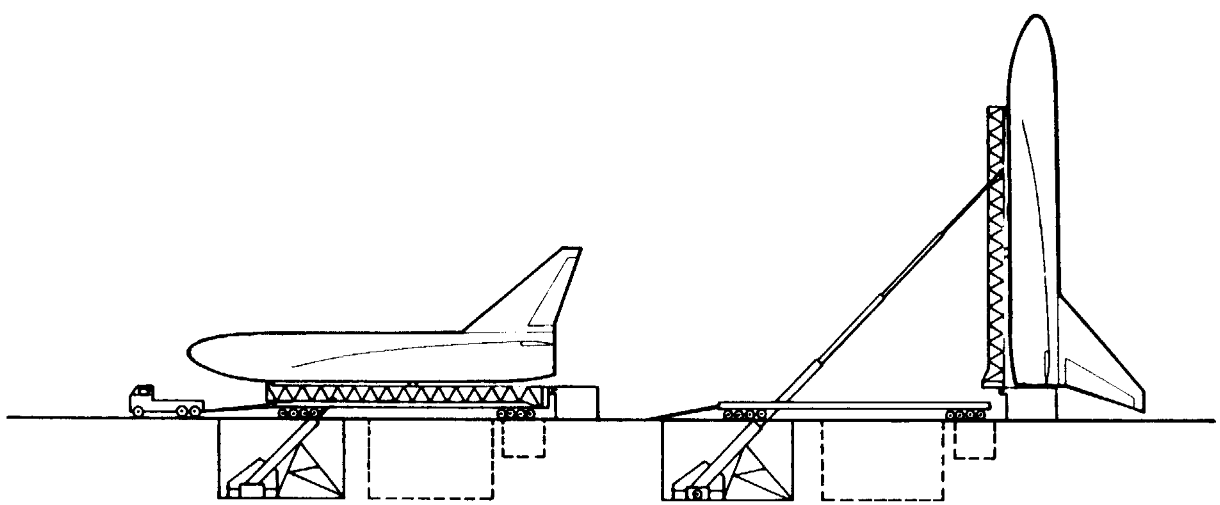


*NOTE:
CRYO SERVICING TIME BASED UPON LOADING BOTH STAGES OF BOTH PROPELLANTS SIMULTANEOUSLY.

PAD ERECTION TECHNIQUES

The Carrier and Orbiter vehicles are individually moved in the horizontal position from the recertification area to the launch pad area. First the Carrier is backed into position adjacent to the launch mount. The tow vehicle is disconnected and the erection device(s) is connected to the vehicle carriage. The device then erects the Carrier and holddown connections are made. The Orbiter now is ready for erection. The orbiter is then brought in nose first over the orbiter erection device(s) and is erected. The device is retracted and the orbiter is moved adjacent to the Carrier and mated. The final step is the emplacement of two service towers for system and servicing connections as well as crew loading. These towers would additionally provide rapid crew egress.

PAD ERECTION TECHNIQUES



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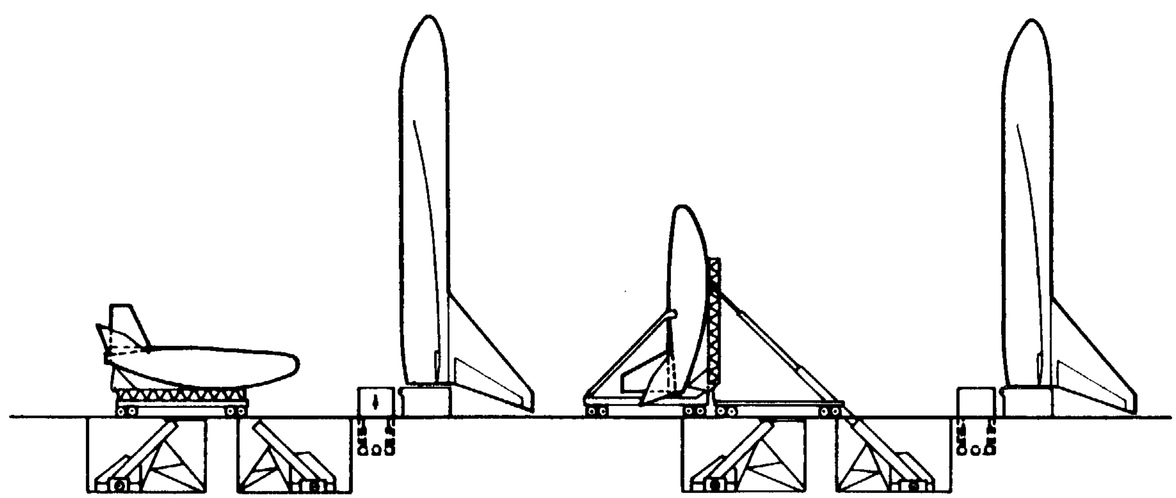
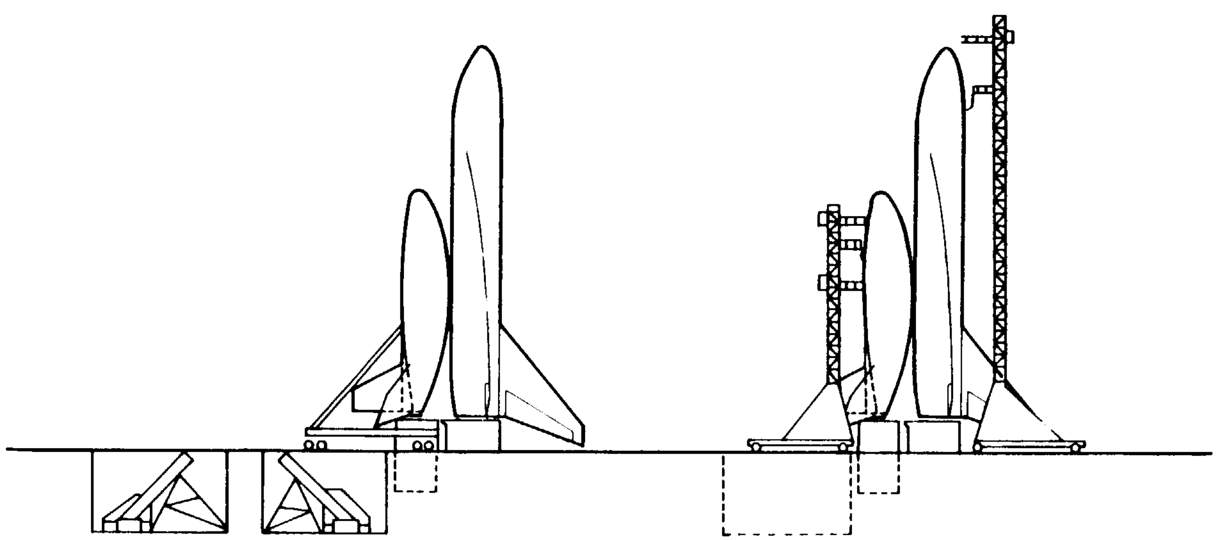
PAD ERECTION TECHNIQUES
(Continued)



ILRVS-282A

PAD ERECTION TECHNIQUES
(Continued)

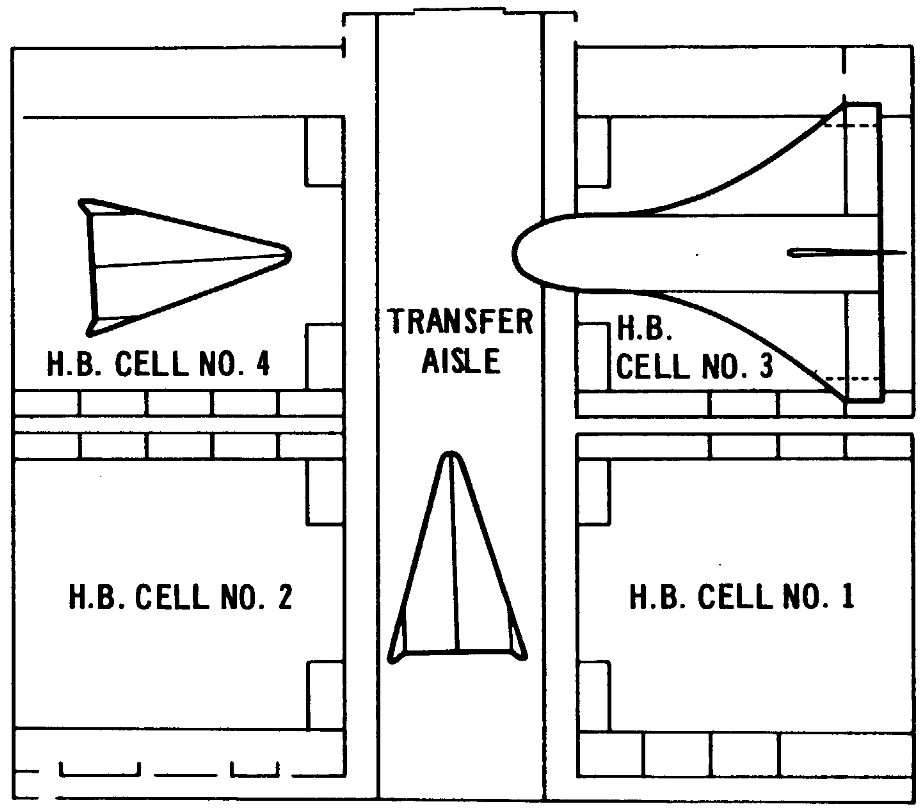


ILRVS-283B

VAB UTILIZATION

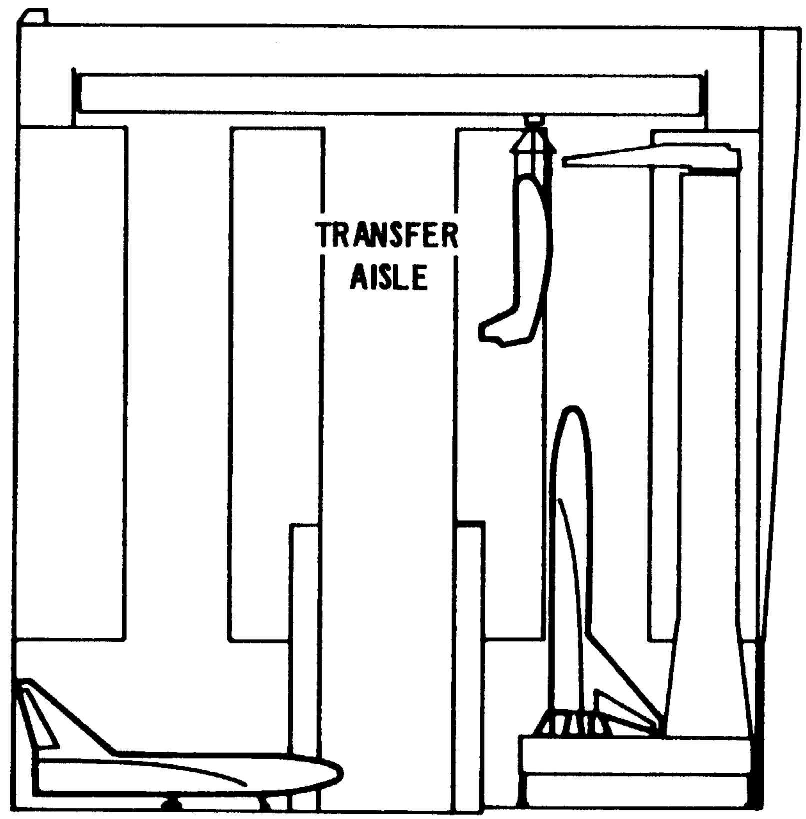
For this concept the Carrier and Orbiter are horizontal and have proceeded through build-up and checkout individually. The Carrier is the first element to be erected. First the Carrier is backed out of the cell and the lift crane is attached. The tail section is attached to a dolly. As the vehicle is erected the dolly moves forward into the cell. When fully erected the dolly is disconnected and the carrier is elevated. As the Carrier is elevated vertically the Launcher unit is moved into the cell. With the Launcher in position the vehicle is lowered onto the launch mount and secured. Concurrent with this activity the Orbiter unit has been loaded and positioned in an adjacent area, ready for vertical erection. This element is erected, moved to the Carrier, lowered, and mating is accomplished. The complete vehicle is now ready for integrated checkout and/or movement to the launch pad.

VAB UTILIZATION
Plan View – Vehicle Horizontal



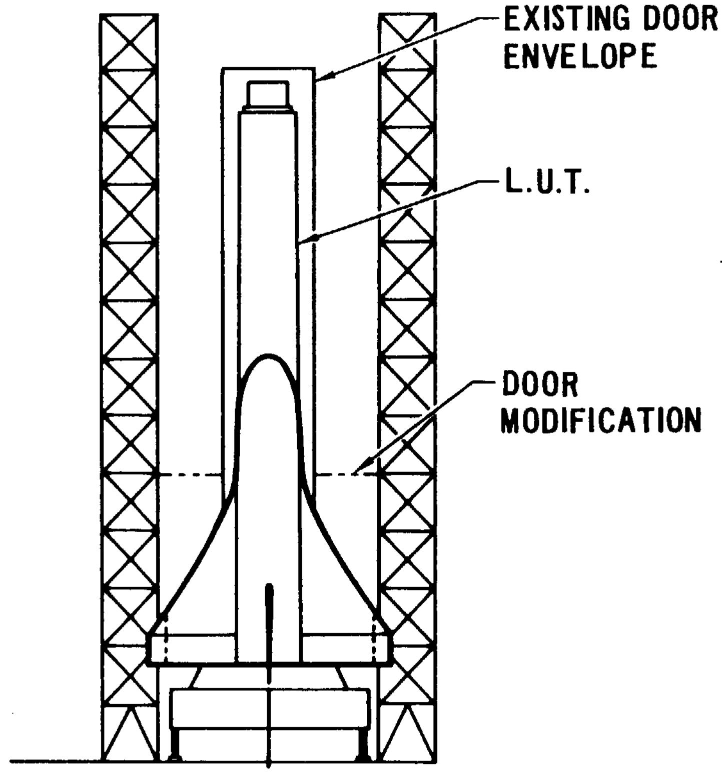
ILRVS-312F

VAB UTILIZATION
Side View



ILRVS-315F

VAB UTILIZATION
High Bay – Side View



ILRVS-314F

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SCHEDULES

AND

COSTS

Task 5

ILRVS-433F

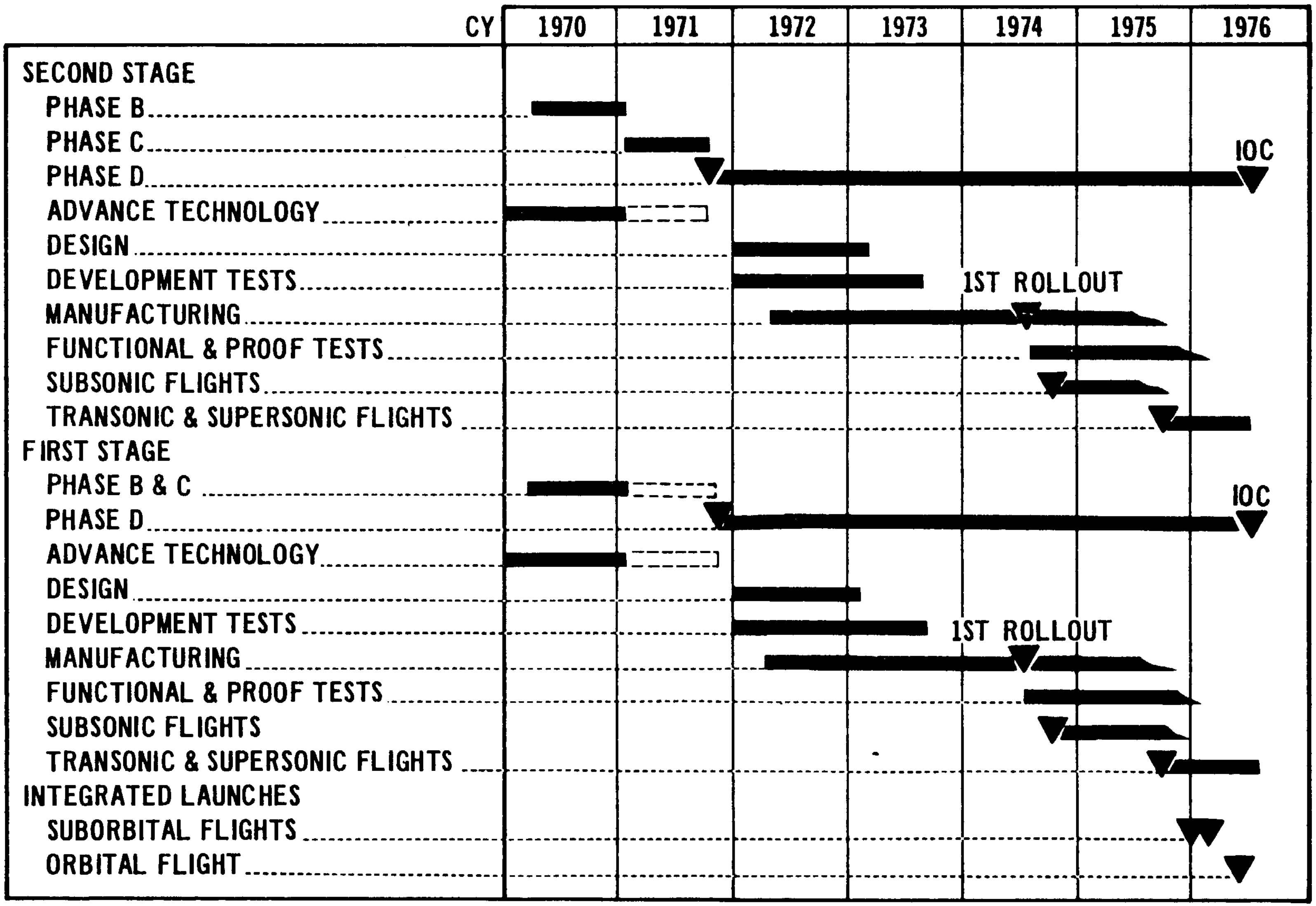
SUMMARY PROGRAM SCHEDULE

The Summary Program Schedule shows the schedule increments from the beginning of Phase B through horizontal and vertical flight test for each stage and the integrated flights including sub-orbital and orbital.

Completion of technology development of several systems to a degree that will permit an unimpeded Phase C and provide a firm basis for Phase D will pace the program. The dotted line indicates possible supporting research and technology development to the beginning of Phase D. Technology development carried beyond that point increases the risk factor.

Key program milestones include rollout at 33 months, first subsonic (horizontal) flight at 35 months, first transonic (vertical) flight 46 months, and first orbital flight at 54 months after Phase D ATP.

SUMMARY PROGRAM SCHEDULE



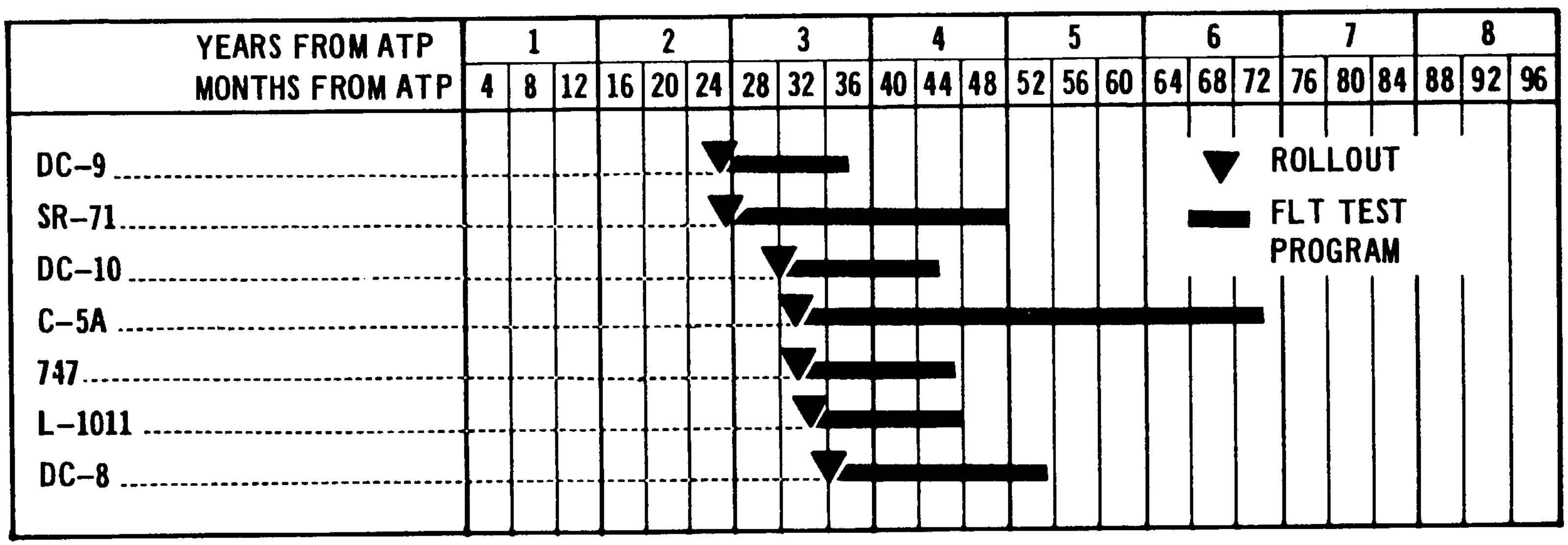
ILRVS-269F

AIRCRAFT DEVELOPMENT SUMMARY

The Aircraft Development Summary is presented as a comparison aid and was used as a baseline for establishing some of the key dates in the Summary Program Schedule.

Activities such as proof tests, functional and integration tests between rollout and first flight are not shown. The range in actual time from one month on the DC-9 to 5 months on the 747, depending on the open items remaining at rollout and the ground test philosophy employed.

AIRCRAFT DEVELOPMENT SUMMARY



FLIGHT TEST APPROACHES

The Flight Test Program will use operational flight test units. These will be full scale production design vehicles with production subsystems installed (subsonic flight test article need not have production heat protection shingles installed). This approach is recommended because it does not incur the additional costs of protype vehicles which could not be used to evaluate many areas of the flight envelope without modification. With this concept the advantages of having only one design and development program are retained.

FLIGHT TEST APPROACHES

- BASIC RESEARCH/SCALE PROTOTYPE
  (M–2, HL–10, X–15)
- FULL SCALE PROTOTYPE – NO PRODUCTION COMMITMENT
  (B–70, YF–12)
- √ OPERATIONAL VEHICLES
  (C–5, GEMINI, APOLLO, COMMERCIAL TRANSPORTS)

INCREASED

- DESIGN & DEVELOPMENT CONSERVATISM
- TIME TO OPERATIONAL PROGRAM
- PROGRAM COST

ILRVS–258F

COST DATA SOURCES

The Cost Estimating Relationships (CER's) that make up the MDAC cost model are based on cost data from Mercury, Gemini, ASSET, BGRV, Saturn S-IVB Stage, F-4 Phantom II Fighter Aircraft, vendor subsystems data, and Industry published cost models. The model was developed under contract to NASA-AMES (Optimized Cost/Performance Design Methodology Study NAS2-5022, 1969).

COST DATA SOURCES

MANNED SPACECRAFT
- MERCURY
- GEMINI
- MOL

UNMANNED SPACECRAFT
- ASSET
- BGRV

LAUNCH VEHICLES
- SATURN S–IVB

AIRCRAFT
- F–4

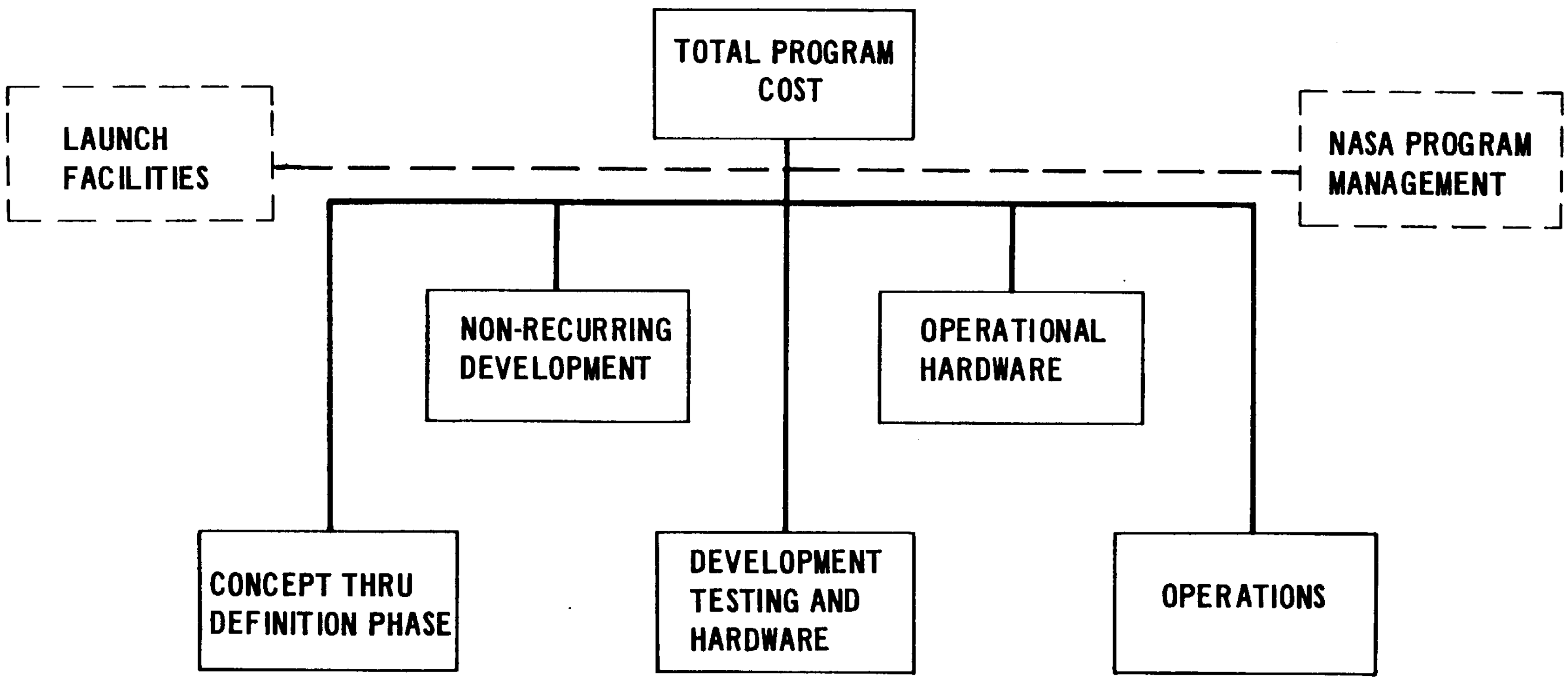
VENDORS
- SUBSYSTEM DATA

ILRVS–365F

TOTAL PROGRAM COST ELEMENT STRUCTURE

- o Concept Thru Definition Phase - Corresponds to phased project planning, Phase B, conducted by several contractors to select a best concept and define preliminary specifications, schedules and plans.
- o Non-Recurring Development - Corresponds to Phases C and D and includes design, development, ground test hardware, and ground testing.
- o Development Testing and Hardware - Corresponds to Phases C and D and includes all flight test hardware and development flight testing.
- o Operational Hardware - Includes acquisiton of all hardware items required to support the op operational phase of the program, and corresponds, in part, to the manufacture function in Phase D.
- o Operations - Includes all recurring labor and material required to support the flight operations from Initial Operational Capability (IOC) through program completion.
- o Launch Facilities - All program peculiar buildings and support installations required to support the flight operations.
- o NASA Program Management - Includes NASA Center Program Office management and system integration activities during the several program phases.

TOTAL PROGRAM COST ELEMENT STRUCTURE

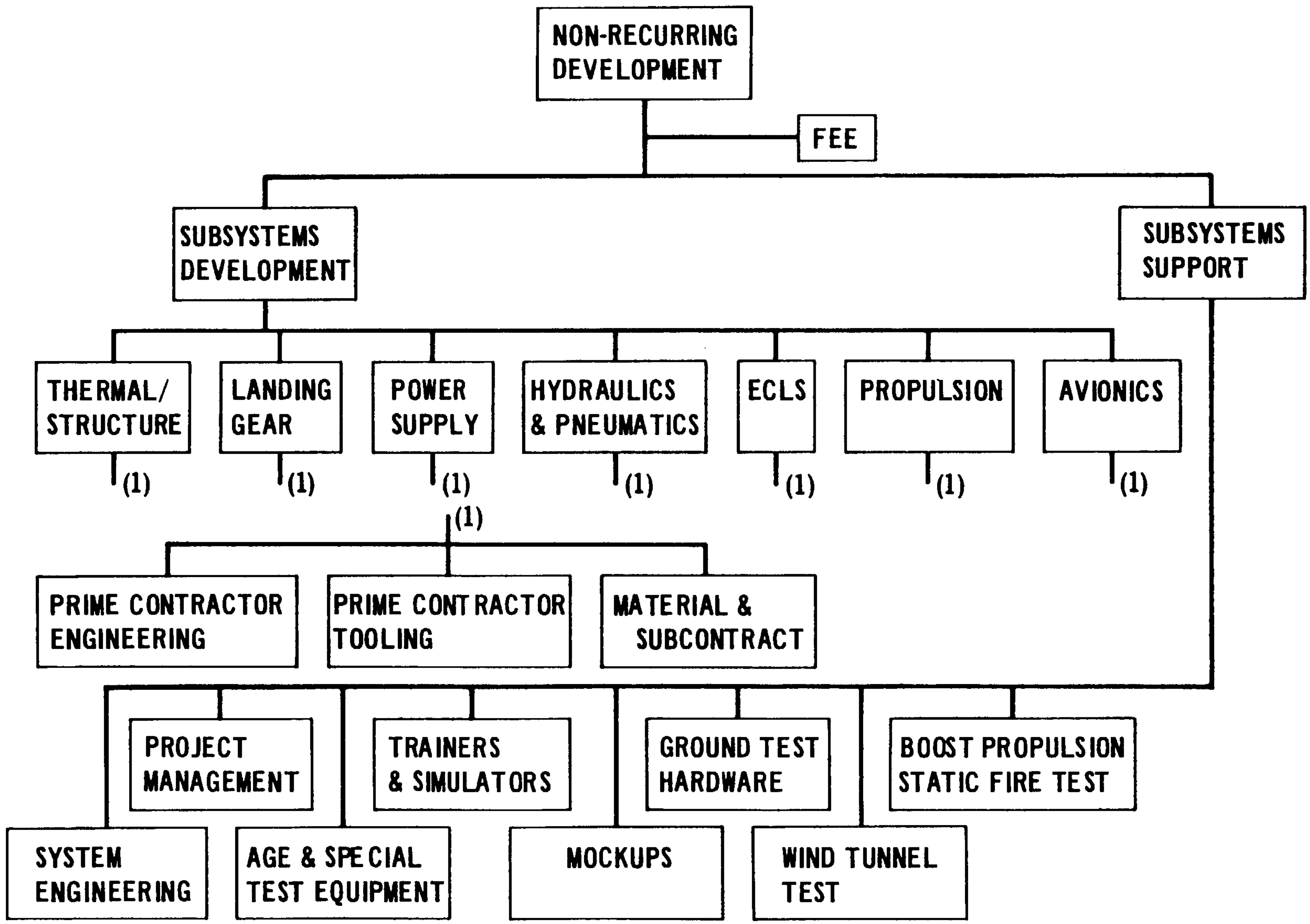


ILRVS-366F

NON-RECURRING DEVELOPMENT COST ELEMENT STRUCTURE

Non-recurring development corresponds to Phases C and D and includes design, development, ground test hardware, and ground testing. Outlined here are the major cost elements contributing to the development cost.

NON-RECURRING DEVELOPMENT COST ELEMENT STRUCTURE

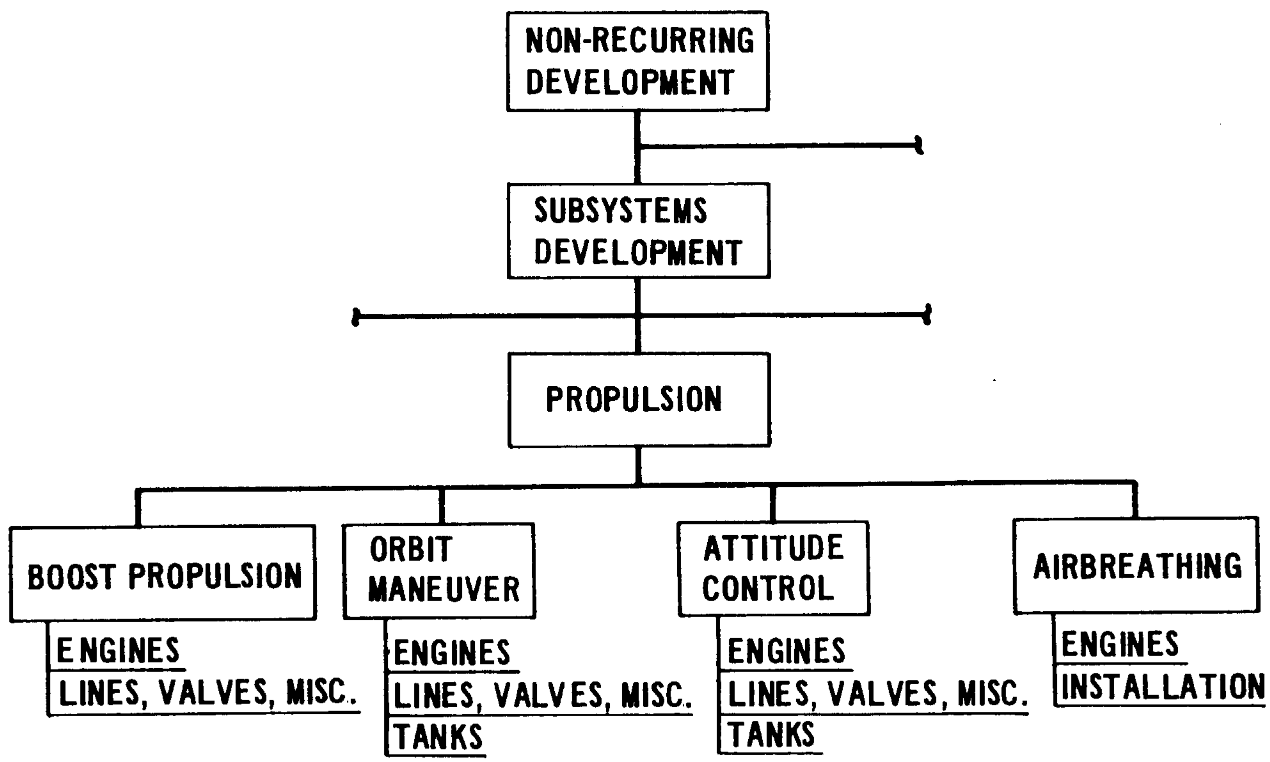


ILRVS-367 F

NON-RECURRING SUBSYSTEM COST ELEMENT STRUCTURE

A breakdown of the propulsion subsystems is presented to indicate the level of detail at which the propulsion cost estimates have been prepared. In general, the level of subsystem breakdown is dependent on cost data availability and the subsystem component sensitivity to the estimating parameters utilized.

NON-RECURRING SUBSYSTEM COST ELEMENT STRUCTURE



ILRVS-368 F

PROGRAMMATIC GROUND RULES

The programmatic considerations shown are for the most part taken from the program study outline (PSO). Payload to orbit delivery probability was computed and is used as the base for computing the actual number of flights required to fulfill a specified mission requirement.

These considerations were used to compute the baseline vehicle requirements and program costs and were used as the base around which the sensitivity analyses were made.

PROGRAMMATIC GROUND RULES

- ALL COSTS ARE IN 1969 DOLLARS
- PAYLOAD COSTS NOT INCLUDED
- ONE PRIME CONTRACTOR FOR COMPLETE SHUTTLE AND ONE FOR ENGINES
- COMMONALITY BETWEEN STAGES FOR SIMILAR SUBSYSTEMS ACCOUNTED FOR IN DEVELOPMENT COSTS
- PAYLOAD TO ORBIT DELIVERY PROBABILITY = 0.97
- 10 YEAR OPERATIONAL PROGRAM
- REFERENCE DESIGN LIFE = 100 FLIGHTS
- REFERENCE MISSION PAYLOAD, 25,000 LB
- REFERENCE LAUNCH RATE, 12 PER YEAR
- LAUNCH, LANDING AND RECERTIFICATION SITE AT SAME LOCATION

ILRVS-249F

DEVELOPMENT COST SUMMARY

Summarized here are those cost elements considered to be chargeable to the development of the total system. Costs included are contractor non-recurring and recurring, and NASA Center costs, and correspond to the top level non-operational elements of the total program cost element structure presented earlier. This is approximately a 6 billion dollar development program and the total costs are approximately equal for both stages.

DEVELOPMENT COST SUMMARY
Millions of 1969 Dollars

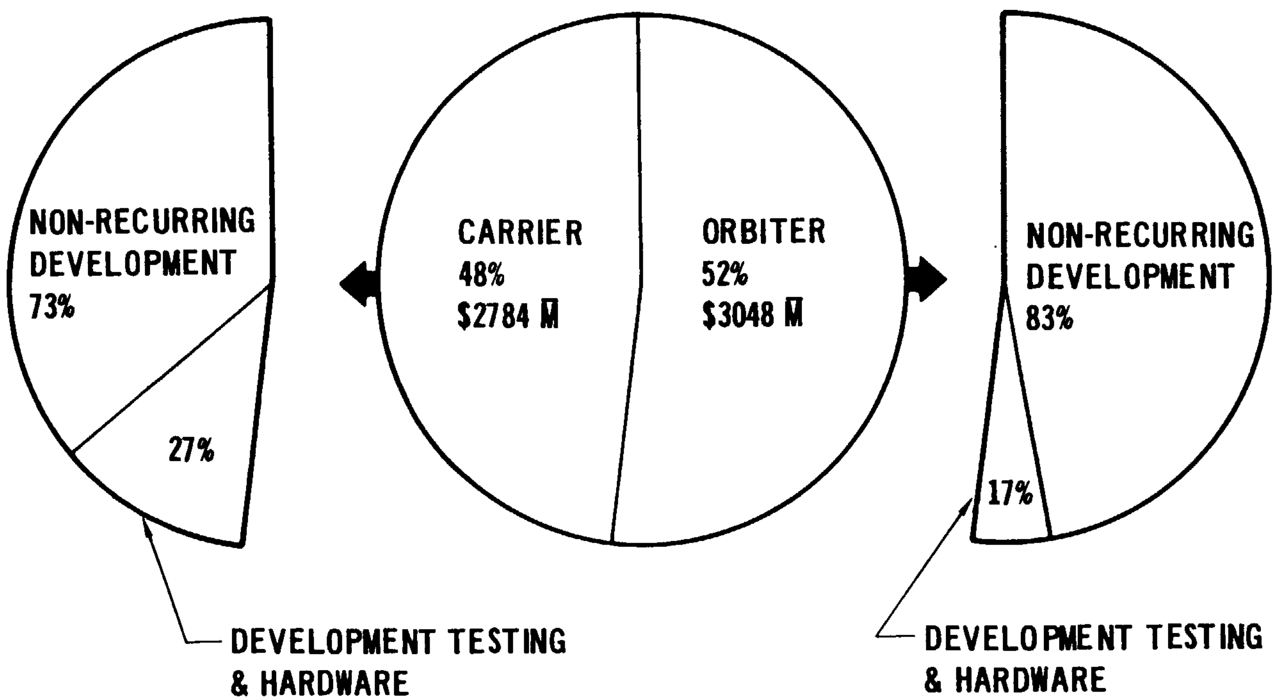
	CARRIER	ORBITER	TOTAL
• CONCEPT THRU DEFINITION PHASE	15	15	30
• NON-RECURRING DEVELOPMENT	2012	2530	4542
• DEVELOPMENT TESTING AND HARDWARE	757	503	1260
SUBTOTAL CONTRACTOR COST	2784	3048	5832
• NASA PROGRAM MANAGEMENT	30	34	64
• LAUNCH FACILITIES	30	20	50
SUBTOTAL	60	54	114
• TOTAL DEVELOPMENT PROGRAM COST	2844	3102	5946

ILRVS-369F

DEVELOPMENT COST DISTRIBUTION

Development cost distribution is affected by the commonality of the subsystem. Although the carrier is the larger vehicle it contributes only 48% of the development cost. The major portion of the development costs for both the carrier and orbiter are attributable to the individual subsystem and subsystem support activities. The primary development costs are charged to the orbiter and small additional costs are charged to the carrier for required modification and integration. This accounts for the relatively lower carrier costs.

DEVELOPMENT COST DISTRIBUTION



ILRVS-370F

NON-RECURRING DEVELOPMENT COST SUMMARY

Non-recurring subsystems development and support costs are given by the major cost elements for both the carrier and orbiter. All costs are in 1969 dollars. Non-recurring development costs reflect commonality of subsystems with the primary development cost charged to the orbiter and a small additional cost charged to the carrier for modifications and peculiarities. Examples of significant cost savings due to commonality are evidenced in the avionics and boost propulsion subsystem. The thermal/structure subsystems, however, reflect independent development costs with minimum savings due to commonality.

NONRECURRING DEVELOPMENT COST SUMMARY

Millions of 1969 Dollars

DEVELOPMENT PROGRAM	CARRIER	ORBITER	TOTAL
NONRECURRING COST			
SUBSYSTEMS DEVELOPMENT			
THERMAL/STRUCTURE	480	297	777
LANDING GEAR	22	10	32
POWER SUPPLY	17	37	54
HYDRAULICS AND PNEUMATICS	18	6	24
ECLS	17	30	47
AVIONICS	85	510	595
BOOST PROPULSION	114	513	627
ORBIT MANEUVER PROPULSION	–	26	26
ATTITUDE CONTROL PROPULSION	18	135	153
AIRBREATHING PROPULSION	133	76	209
SUBTOTAL	904	1640	2544
SUBSYSTEMS SUPPORT			
SYSTEM ENGINEERING	108	85	193
PROJECT MANAGEMENT	37	26	63
AGE AND SPECIAL TEST EQUIPMENT	247	210	457
TRAINERS AND SIMULATORS	115	96	211
MOCKUPS	23	14	37
GROUND TEST HARDWARE	306	184	490
WIND TUNNEL TEST	10	6	16
BOOST PROPULSION STATIC FIRE TEST	79	39	118
SUBTOTAL	925	660	1585
SUBTOTAL NONRECURRING COST	1829	2300	4129
FEE AT 10%	183	230	413
TOTAL NONRECURRING COST	2012	2530	4542

ILRVS-371F

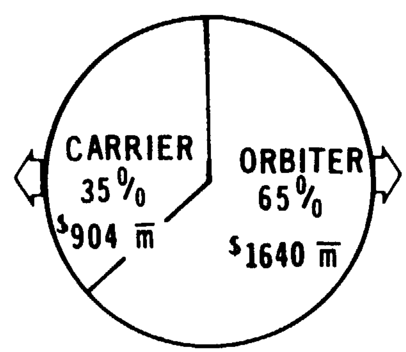
NON-RECURRING SUBSYSTEMS DEVELOPMENT COST DISTRIBUTION

Since the development costs for common subsystems are charged primarily to the orbiter, the orbiter accounts for two thirds of total development costs. With this method of bookkeeping, the thermal structure is the major subsystem cost for the carrier, and the avionics and boost propulsion subsystems are the dominant costs for the orbiter.

NONRECURRING SUBSYSTEMS DEVELOPMENT COST DISTRIBUTION

CARRIER

SUBSYSTEM	PERCENT
THERMAL/STRUCTURE	53.1
LANDING GEAR	2.4
POWER SUPPLY	1.9
HYDRAULICS & PNEUMATICS	2.0
ECLS	1.9
AVIONICS	9.4
BOOST PROPULSION	12.6
ATTITUDE CONTROL PROPULSION	2.0
AIRBREATHING PROPULSION	14.7



ORBITER

PERCENT	SUBSYSTEM
18.1	THERMAL/STRUCTURE
.6	LANDING GEAR
2.3	POWER SUPPLY
.4	HYDRAULICS & P NEUMATICS
1.8	ECLS
31.1	AVIONICS
31.3	BOOST PROPULSION
1.6	ORBIT MANEUVER PROPULSION
8.2	ATTITUDE CONTROL PROFULSION
4.6	AIRBREATHING PROPULSION

DEVELOPMENT TESTING AND HARDWARE

The development test program consists of a horizontal take-off and landing subsonic test program and a vertical suborbital and orbital test program. Included are two complete production hardware articles. The horizontal flight test program consists of 140 flights on the orbiter and 140 flights on the carrier. The vertical flight test program consists of 6 flights on the orbiter, 6 flights on the carrier, and 3 combined flights. Refurbishment costs are for repairs and modifications resulting from the test program to maintain and return the vehicles to an operational status.

DEVELOPMENT TESTING AND HARDWARE
Millions of 1969 Dollars

	CARRIER	ORBITER	TOTAL
HORIZONTAL FLIGHT TESTING	30	21	51
VERTICAL FLIGHT TESTING	77	87	164
FLIGHT TEST HARDWARE	440	255	695
SPARES	54	31	85
REFURBISHMENT	87	63	150
SUBTOTAL	688	457	1145
FEE AT 10%	69	46	115
TOTAL DEVELOPMENT TESTING AND HARDWARE	757	503	1260

ILRVS-373F

SPACECRAFT INVENTORY

The effect of increasing the basic mission from 7 days to 30 days on the Orbiter inventory is shown in the accompanying table. In addition, the effect of launch rate on spacecraft inventory is given. From the table it is seen that, for a 10-year program,with the probabilities, reliabilities, and design life as stated, the Carrier inventory varies from 2 vehicles at the low (4 launches per year, nominal) launch rate to 15 vehicles at the high (100 launches per year, nominal) launch rate. At the same time, the Orbiter inventory varies from 2 vehicles to 19 vehicles over the same range of launch rates and for the nominal 7-day mission. For a 30-day mission, the upper limit is extended to 21 vehicles for the Orbiter.

SPACECRAFT INVENTORY

- 10-YEAR PROGRAM
- DESIGN LIFE = 100 USES
- 1-DAY CARRIER MISSION
- PAYLOAD TO ORBIT DELIVERY RELIABILITY = 0.97
- P_{SR} (ORBITER) = .99
- P_{SR} (CARRIER) = .995

ANNUAL LAUNCH RATE	NUMBER OF VEHICLES REQUIRED		
	CARRIER	ORBITER	
		7-DAY MISSION	30-DAY MISSION
4	2	2	2
8	2	2	2
10	2	2	3
12 REFERENCE	2	3	3
30	4	6	7
50	7	10	11
100	15	19	21

ILRVS-245F

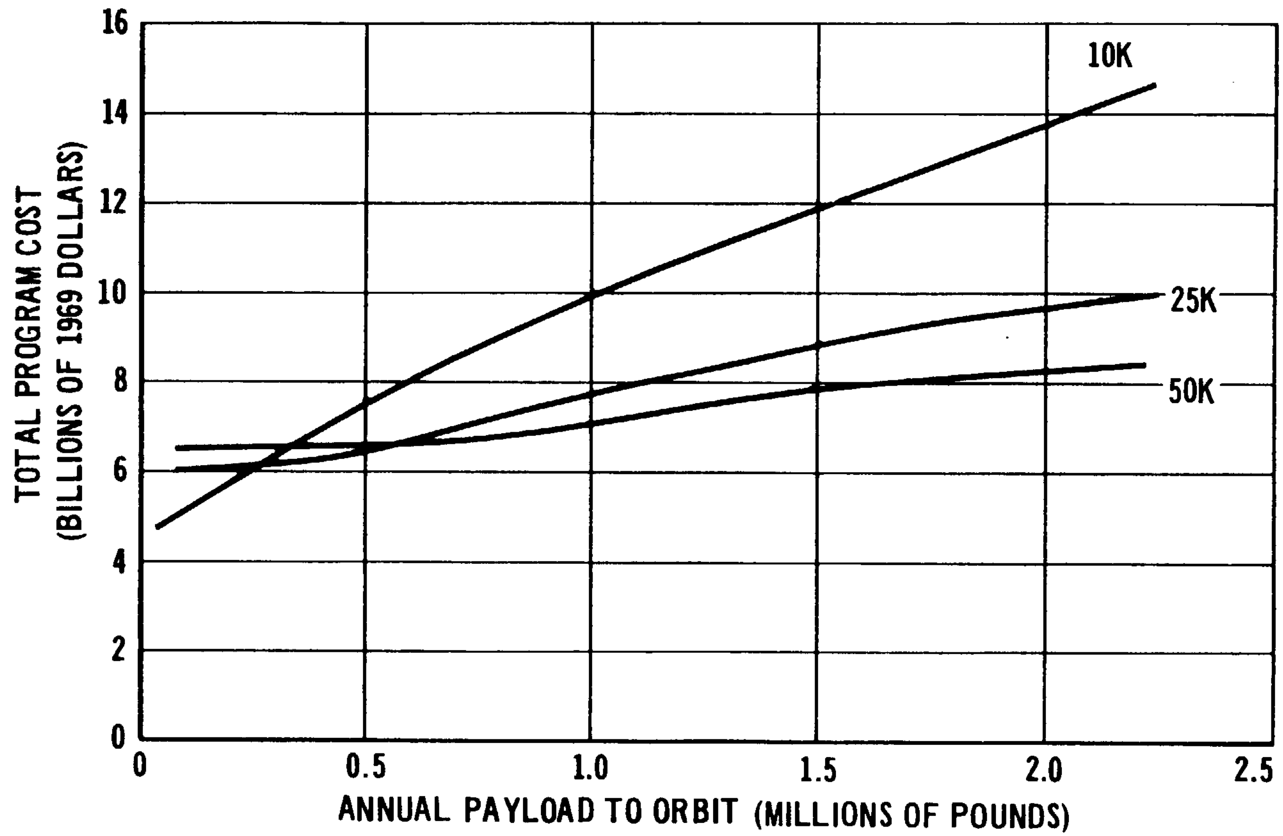
EFFECT OF PAYLOAD CAPABILITY AND ANNUAL TRANSPORT DEMAND

As the annual or total payload to orbit increase beyond about 200,000 pounds annually, there is a definite cost advantage to the use of larger vehicles capable of delivering 25,000 to 50,000 pounds of cargo on each flight. Below 200,000 pounds annually, there is a slight advantage to use of a smaller vehicle of 10,000 pound payload capability.

At the larger annual cargo rates, the cost of flight hardware becomes the large driving cost for small vehicles. For the 10,000 pound vehicle and a million pounds of cargo annually, the development cost and the flight hardware costs are about equal at $5.40 and $3.44 billions respectively while the operational costs are about $940 millions. At two million pounds annually, the flight hardware will cost $6.64 billions and operations $1.7 billions.

The larger vehicles cost slightly more to develop but reduce total costs by smaller quantities of flight hardware and lower operations costs. At two million pounds annually for the 25,000 pound vehicle, the flight hardware costs $2.87 billions and operations $842 millions. The 50,000 pound vehicle flight hardware would cost $1.43 billions and operations $543 millions.

EFFECTS OF PAYLOAD CAPABILITY AND ANNUAL TRANSPORT DEMAND



ILRVS-382F

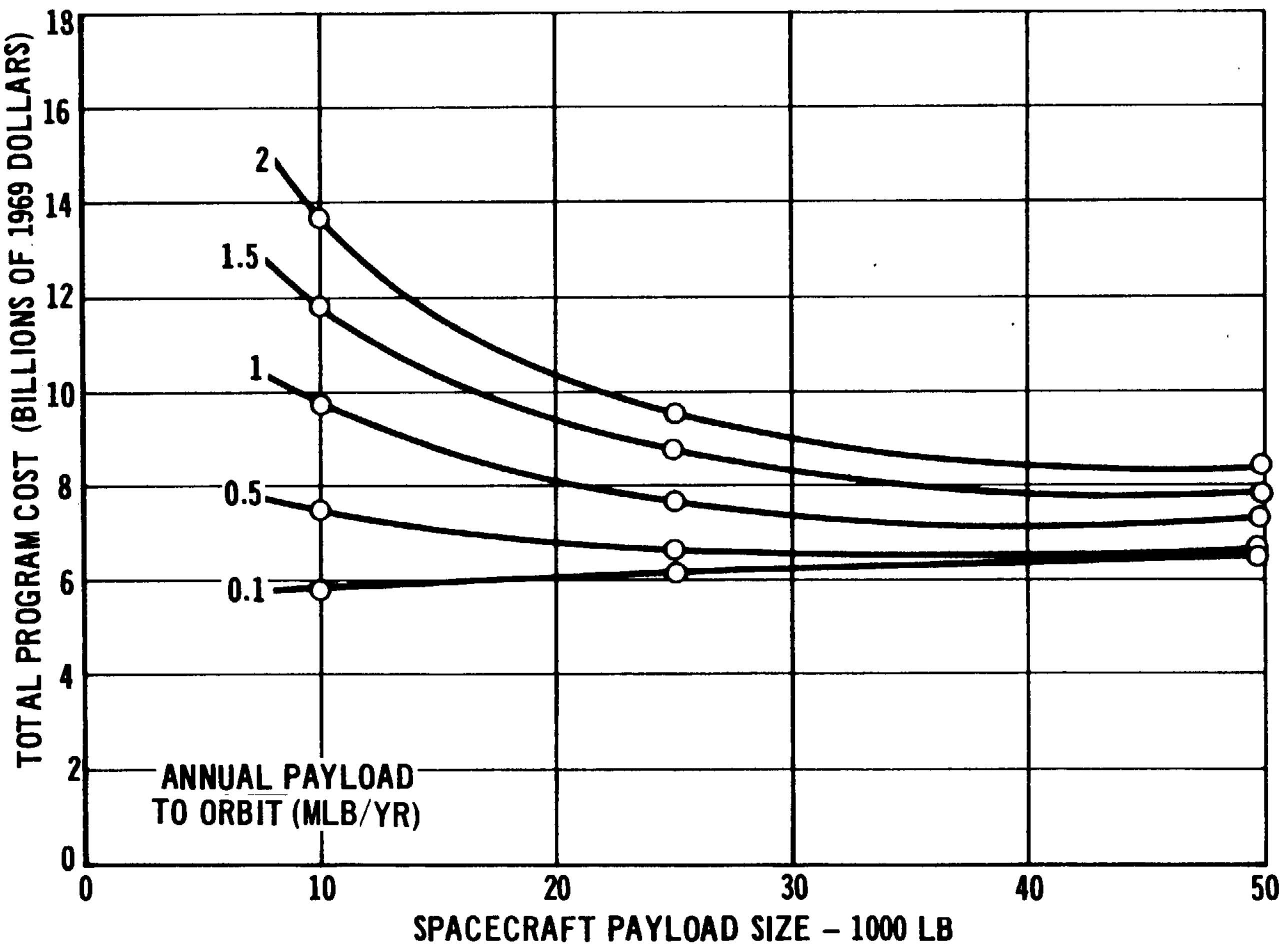
EFFECT OF PAYLOAD CAPABILITY AND ANNUAL TRANSPORT DEMAND (10 Year)

For minimum total costs, the logistics vehicle should be tailored to the cargo delivered-to-orbit requirements. When total cargo is undefined, some judgement must be made concerning a desirable size vehicle. This figure indicates that designing for large payloads is desirable. For low traffic rates it will increase total costs slightly, but for high traffic will significantly reduce total costs.

At larger payloads, these curves show total program cost increases. In other words, these curves would all indicate buckets or minimum at some specific cargo size. As indicated, the costs are rather insensitive to the specific cargo size. This suggests that it is better to design for big cargo capability and back-off if necessary in order to hold total costs down as the design evolves into hardware.

EFFECTS OF PAYLOAD CAPABILITY AND ANNUAL TRANSPORT DEMAND

10-Year Programs



ILRVS-381F

PROGRAM COST SUMMARY

A summary of the Program Costs for the 25,000 pound discretionary payload spacecraft is shown on the opposite page for three traffic rates; four, 12 and 100 successful flights per year.

The total development cost is the same for all cases. The additional operational hardware required is that flight hardware above that required for the flight test program to fulfill the requirements for the number of successful launches specified. The factors used in determining the number of flights and vehicles required are as stated in the programmatic ground rules plus the launch to launch probability of the orbiter and the carrier at .99 and 995 respectively.

Included at the bottom of the chart are the average operational cost per launch and the average operational cost per pound of discretionary payload delivered to orbit. The primary factors decreasing these costs with increasing number of launches are the more efficient use of the manpower required to support the launch rate and the expected decrease in the amount of sustaining material.

PROGRAM COST SUMMARY
25,000 Pound Payload Vehicle Millions of 1969 Dollars

COST CATEGORY \ TRAFFIC RATE	LOW (4 FLIGHTS/ YEAR)	REFERENCE (12 FLIGHTS/ YEAR)	HIGH (100 FLIGHTS/ YEAR)
NON-RECURRING	4542	4542	4542
DEVELOPMENT TEST PROGRAM	1260	1260	1260
TOTAL DEVELOPMENT	5802	5802	5802
ADDITIONAL OPERATIONAL HARDWARE	–	110	3507
OPERATIONS COST	206	357	1682
TOTAL PROGRAM	6008	6269	10991
AVERAGE OPERATIONAL COST/FLIGHT	5.15	2.97	1.68
AVERAGE OPERATIONAL COST/POUND OF DISCRETIONARY PAYLOAD TO ORBIT	$206/POUND	$119/POUND	$67/POUND

ILRVS-380F

TRANSPORT COST EVALUATION

The Program Study Outline (PSO) specified that the Saturn IB/CSM (AAP Configuration) should be used as the base from which to compare the payload to orbit costs of the ILRVS system to determine if an order of magnitude reduction in cost has been achieved. This table presents the required comparative evaluation summary. Greater than an order of magnitude has been achieved in both cases on the same ground rules. Lofted discretionary payload costs were reduced from $30,000 per pound to $900 per pound and total lifted payload from $4,000 per pound to $92 per pound.

These reductions are primarily due to the reusability and the desinged in potential of low launch times and costs.

TRANSPORT COST EVALUATION

AN ORDER OF MAGNITUDE REDUCTION IN RECURRING COST HAS BEEN INDICATED.

	PROGRAM STUDY(1) OUTLINE INDEX	MDAC–ILRS(2) STUDY RESULT
LOFTED DISCRETIONARY PAYLOAD	$30,000/POUND	$900/POUND
TOTAL LOFTED PAYLOAD (INJECTED WEIGHT)	$ 4,000/POUND	$ 92/POUND

(1) SATURN 1B/CSM (AAP CONFIGURATION), 4 PER YEAR LAUNCH RATE

(2) AVERAGE COST FOR 10 YEAR PROGRAM, 4 PER YEAR LAUNCH RATE, 25,000 POUND PAYLOAD VEHICLE

ILRVS-348

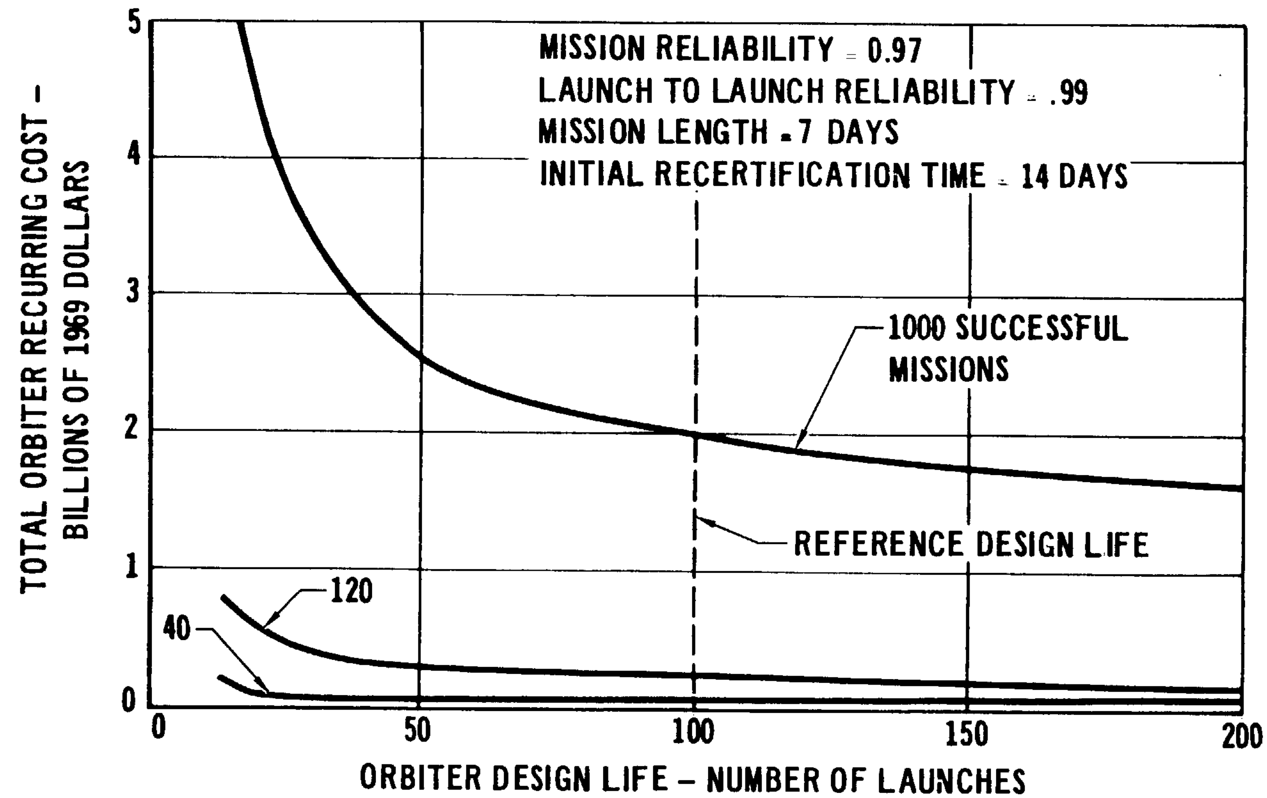
COST/DESIGN LIFE INTERACTION
ORBITER

The sensitivity of total orbiter recurring cost to orbiter design life is shown. The reference orbiter design life is 100 launches. The cost effect on programs of 1000, 120 and 40 successful missions is shown. The recurring cost includes investment hardware, initial spares, sustaining spares, operations and recertification. The cost of the two RDT&E orbiters and their initial spares which are used in the operational phase are not included in the recurring cost.

As the orbiter design life increases, the total recurring cost decreases. At low design life values, the orbiter inventory quantities are high and the corresponding recurring costs are high. As the design life increases, the inventory quantities decrease rapidly at first and the recurring costs also decrease. The decrease in inventory and recurring cost slows as the design life becomes large. This slowdown reflects the fact that the vehicle design life is no longer a critical factor in orbiter inventory determination. Consequently, for each program illustrated the sensitivity of recurring cost to orbiter design life diminishes and can be disregarded after a sufficiently high design life has been achieved.

For the 40 successful mission program, a minimum inventory and recurring cost are achieved with a design life of 25 launches. For the 120 successful mission program, the reference design life of 100 launches represents a minimum cost system. For the 1000 successful mission program, a design life of around 200 missions is appropriate.

COST/DESIGN-LIFE INTERACTION (ORBITER)



ILRVS-276F

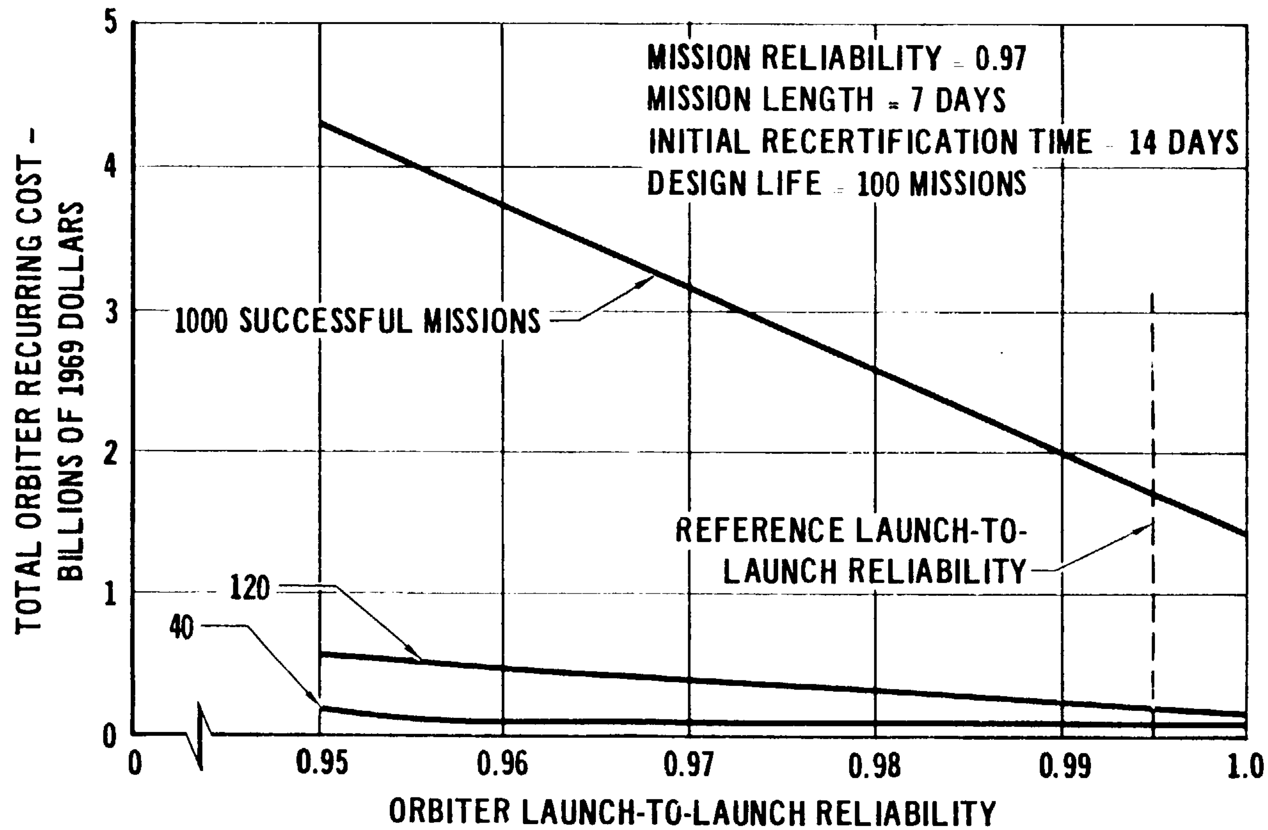
COST/RELIABILITY INTERACTION
ORBITER

The sensitivity of total orbiter recurring cost to launch-to-launch reliability is shown. Programs of 40, 120 and 1000 successful missions for a ten year operational phase are shown. The reference launch to launch reliability is .995 and the recurring cost includes investment hardware, initial spares, sustaining spares, operations and recertification. The cost for the two RDT&E orbiters and their initial spares which are used for the operational phase are not included in the recurring cost.

Recurring costs decrease approximately linearly with increasing reliability. The rate of cost change with reliability increases as the number of successful missions increases, e.g. the recurring cost increases 600M dollars for each .01 decrease in reliability for the 1000 successful mission program while the recurring cost for the 120 successful mission program only increases 60M dollars for each .01 decrease in reliability.

The 40 successful mission program reaches a minimum of approximately 80M dollars for reliability greater than .96 because the two RDT&E orbiters can support the entire operational phase.

COST/RELIABILITY INTERACTION (ORBITER)

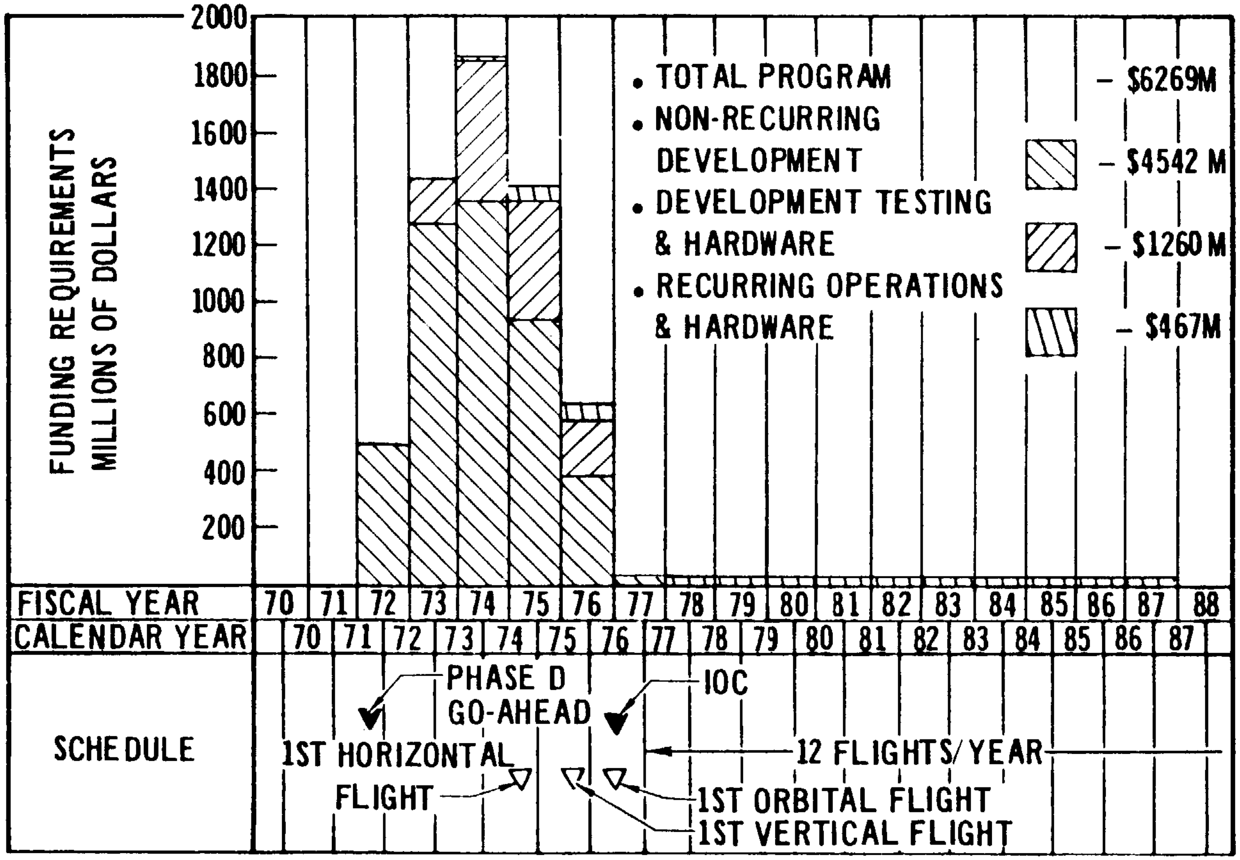


ILRVS-275F

PROGRAM SCHEDULING AND FUNDING
(REFERENCE PROGRAM)

This figure indicates the annual funding requirements for the reference program. Three costs areas are identified: non-recurring development, development testing and hardware, and recurring operations and hardware. The ten-year operational program assumes twelve successful flights per year. Two carriers and three orbiters are required to support this operational program. The two carriers and two orbiters used for the development testing will be used for the operational phase. First orbital flight is scheduled for 54 months after go-ahead. IOC occurs 2 months later.

PROGRAM SCHEDULING AND FUNDING
(Reference Program)



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PROGRAMMATIC CONCLUSIONS

This chart summarizes the conclusions from the programmatic analyses made in support of this study. The order of magnitude reduction in recurring cost over that specified is easily achieved and the most cost effective payload vehicle for various probable annual payload to orbit requirements is defined.

Recertification times up to 50 days do not increase recurring costs for the conditions examined which are given in the programmatic ground rules. The specification of design life for low launch rates is not critical in terms of recurring costs. A design life in the neighborhood of 50 launches is recommended. For high launch rate systems, around 100 launches per year, the design life specification should be increased to around 175 to 200 launches.

In the sensitivity analyses conducted, the overriding parameter is that of probability of successful launch to launch reliability. A change in .01 percent in probability can change the recurring cost around 10 percent and at high launch rates, 100 per year, the cost change is 20 percent.

PROGRAMMATIC CONCLUSIONS

FOR THE VEHICLE PAYLOADS AND MISSION CONDITIONS STUDIED –

- ANALYSIS CONFIRMS POTENTIAL ORDER OF MAGNITUDE REDUCTION IN RECURRING COST
- THE FOLLOWING SPACECRAFT/ANNUAL PAYLOAD TO ORBIT COMBINATIONS ARE THE MOST COST-EFFECTIVE:

AVERAGE ANNUAL PAYLOAD TO ORBIT RANGE	DESIRED SPACECRAFT PAYLOAD
50,000 TO 200,000 POUNDS	10,000 POUNDS
200,000 TO 450,000 POUNDS	25,000 POUNDS
450,000 TO 2,500,000 POUNDS	50,000 POUNDS

- RECERTIFICATION TIME EXCURSIONS UP TO 30 DAYS HAVE LITTLE EFFECT ON OPERATING COSTS.
- DESIGN LIFE SPECIFICATION FOR LAUNCH RATES OF LESS THAN 12 PER YEAR SHOULD BE IN THE 25 TO 100 USE RANGE; ABOVE 12 PER YEAR, IN THE 100 TO 150 USE RANCE.
- LAUNCH-TO-LAUNCH RELIABILITY IS VERY CRITICAL – A CHANGE OF 0.01 CAN INCREASE OR DECREASE THE RECURRING COST BY 20 PERCENT, PARTICULARLY FOR HIGH LAUNCH RATES.

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TECHNOLOGY IDENTIFICATION

Task 7

ILRVS-435F

TECHNOLOGY TREE

The purpose of this chart is to show how the basic performance and operational parameters of the space shuttle interface with the identified technologies and supporting research areas. Some of the technologies have an effect on more than one parameter, i.e., the structure is effected by the thermal protection materials and reusability is influenced by the survivability of these materials. Therefore, some technologies are listed more than once. The chart shows that all phases of the vehicle and its operational aspects are dependent to some degree on the solution of these problem areas.

TECHNOLOGY TREE

- SPACE SHUTTLE
  - PERFORMANCE
    - PROPULSION
      - HIGH P_C ENGINE
      - GASEOUS O_2/H_2 ACS
    - AERO/THERMO
      - CONFIGURATION EVALUATION
      - AERO HEATING
      - STAGE SEPARATION
      - ENERGY MANAGEMENT
      - AUTOMATIC LANDING
    - THERMO-STRUCTURES
      - MATERIALS
      - DESIGN
  - SAFETY/RELIABILITY
    - INTEGRATED AVIONICS
    - JET ENGINE STORAGE
    - STAGE SEPARATION
  - REUSABILITY
    - CRYOGENIC INSULATION
    - MATERIAL SURVIVABILITY
  - TURNAROUND TIME
    - INTEGRATED AVIONICS

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TECHNOLOGY FLOW

This diagram depicts the flow of the identified essential and significant supporting research and technologies into the various phases of the study. Boost engine technology, presently under study in the XLR 129 program, is expected to demonstrate feasibility by the beginning of a Phase D shuttle development program in late 1971.

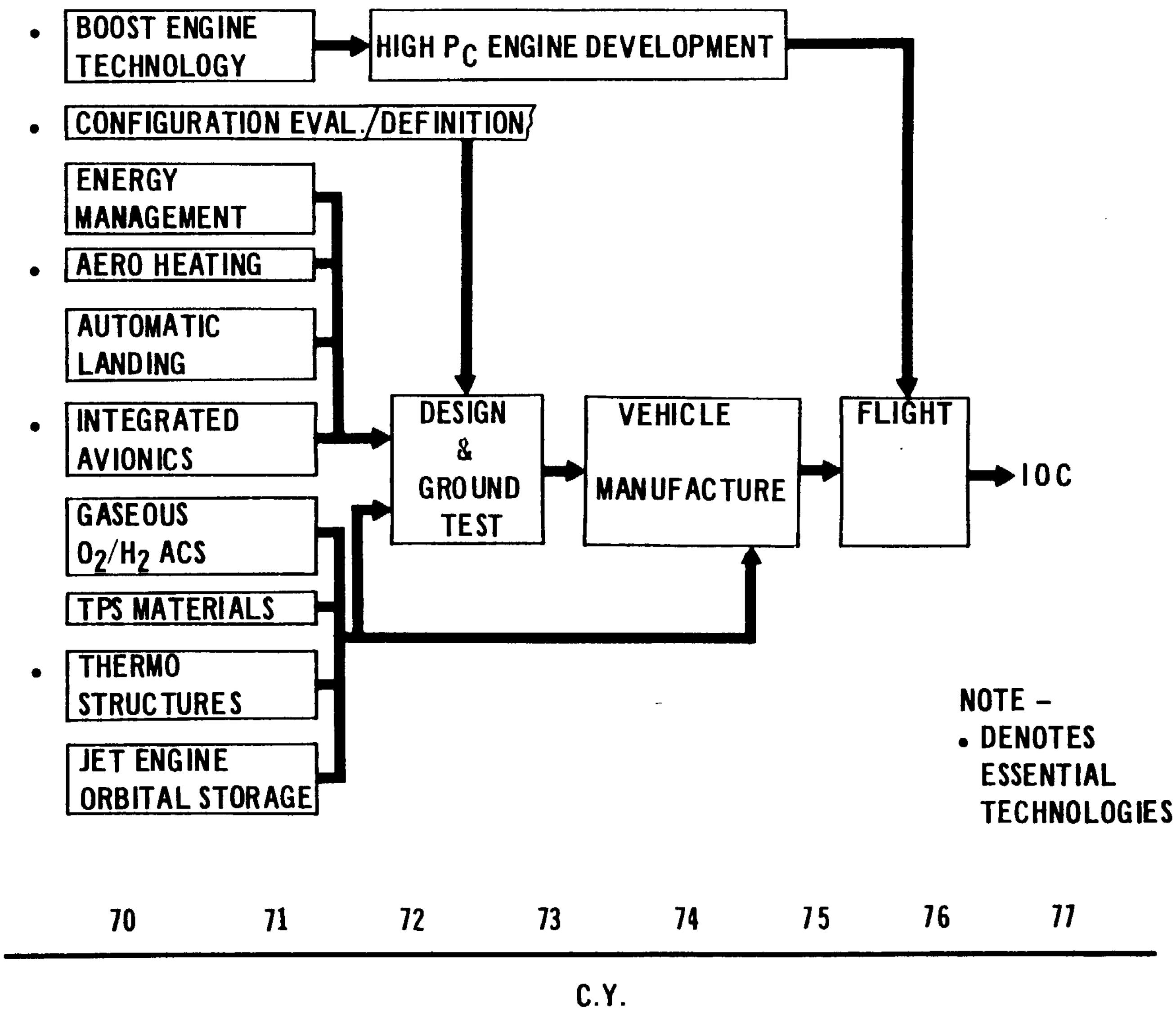
Configuration evaluation must be started at the earliest possible time. The first stage will have to have wind tunnel configuration analysis and definition testing to bring it up to the test level of the HL-10 which has had considerable test hours. It is necessary to start model construction as soon as possible. Configuration evaluation through Phase C will include testing of the launch configuration to investigate interference and stage separation.

Entry energy management, boundary layer transition and automatic landing consist mainly of studies and analysis using wind tunnels, computers and simulators. These studies and integrated avionics must be fed into the design phase early in the study.

Feasibility information from the remaining areas including the attitude control system, thermal protection materials, integral tank design and insulation, must be fed into the design phase with additional impact on manufacturing.

To attain the 1976 IOC date it is necesarry that technology feasibility studies and long lead time developments be initiated as early as possible in order to minimize program risk.

TECHNOLOGY FLOW



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TECHNOLOGY COST SUMMARY

Supporting research and technologies identified during the conceptual study were not sufficiently defined to arrive at good cost figures, thus the wide spread in the estimates.

Configuration evaluation is probably the major technology and will continue through all phases of the design and development effort. The funding shown on this chart is for preliminary wind tunnel testing to study configuration analysis and definition of the first stage vehicle, some heat transfer and hypersonic testing of the HL-10 and tests of the launch configuration to study interfererce and separation problems.

Essential technologies are defined as those required to be solved prior to final design and acquisition and represent high risk areas to the program. Failure of any of these developments could seriously jeopardize the program. The significant research or technologies effect performance or safety to such a degree that the solution of these problems justifies the cost involved.

High costs for the avionics type technologies are attributed to breadboard demonstrations.

TECHNOLOGY COST SUMMARY

TECHNOLOGY		ESTIMATED COST RANGE (THOUSANDS OF DOLLARS)
ESSENTIAL –		
CONFIGURATION EVALUATION		8000 – 9000
AERO HEATING		1500 – 2000
TPS MATERIALS		3500 – 4500
THERMOSTRUCTURES		2000 – 3000
HIGH P_C ENGINE		(1)
SELF TEST FOR ON-BOARD CHECKOUT		3500 – 5000
	SUBTOTAL	18,500 – 23,500
SIGNIFICANT –		
INTEGRATED AVIONICS DEMONSTRATION		4000 – 5000
DATA BUS		1250 – 2000
ELECTRONIC CONTROLS & DISPLAYS		2500 – 4500
NON-COOPERATIVE RENDEZVOUS		500 – 750
GASEOUS O_2/H_2 ACS		3500 – 4500
ENTRY ENERGY MANAGEMENT		400 – 600
AUTOMATIC LANDING		300 – 500
JET ENGINE STORAGE		1500 – 2500
	SUBTOTAL	13,950 – 20,350
NOTE: (1) EXISTING FUNDED PROGRAM	TOTAL	32,450 – 43,850

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SUMMARY

AND

RECOMMENDATIONS

ILRVS-499F

OBSERVATIONS AND FINDINGS

The study has demonstrated that the HL-10/carrier system is feasible. The system is well matched to the requirements. In particular, for the payloads of interest, reasonably high propellant mass fractions are attainable for the HL-10 without ballast penalty.

An adequate technological base exists to assure that the system-essential technology areas can be addressed successfully. The only qualification is that the program proceed in an orderly fashion rather than on a crash basis. This is particularly true for the configuration evaluation effort where a crash program could force a premature decision that might well be costly and result in IOC delay, or even jeopardize the feasibility of the program.

The integral tank concept applied to the HL-10 has demonstrated that the performance benefits overshadow the weight penalties. The result is that the total volume of the vehicle is efficiently utilized.

High thrust-to-weight is beneficial in reducing gravity-induced velocity losses. Hence, some minor base area modification improves the HL-10 performance.

The 7-1/2 years of effort expended in characterizing the HL-10 offers potential R&D benefits. This is particularly true in the aero/thermo testing area.

Unless proper consideration is given to permissible internal environment (cold, warm, hot), the thermo-structures approach can create artificial balance problems and weight penalties.

The idle mode operation of the orbiter boost engines offers benefits in terms of safety and system reliability. However, because of the throttle level (approximately 20%) required to avoid nozzle separation at low altitudes, the attendant penalty in gross lift-off weight (approximately 25%) appears excessive.

Assessment of alternate payloads shows that system weight is more sensitive to payload geometry, particularly diameter, than to payload weight. Large payloads (22'D x 60'L) create large orbiter volumes available for propellant but base areas inadequate for the desired thrust levels. This, coupled with the fact that the same size HL-10 accommodates both or either the 15'D x 60'L and the 22'D x 30'L payloads, suggests that the system performance can be optimized if the payload shape was unconstrained.

Of the dominant system sensitivities listed, the first three are controlled by the operational option chosen. The last two are "facts of life" - all avenues to reducing orbiter inert weights and increasing engine specific impulse must be pursued vigorously.

OBSERVATIONS AND FINDINGS

- HL–10/CARRIER SYSTEM IS FEASIBLE AND WELL MATCHED TO REQUIREMENTS
- TECHNOLOGY STATUS ADEQUATE FOR ORDERLY PROGRAM
- EFFICIENT HL–10 VOLUME UTILIZATION DEMONSTRATED
- MINOR HL–10 BASE AREA MODIFICATIONS DESIRED FOR BETTER PERFORMANCE
- HL–10 STATUS SAVES – R&D COST (AERO/THERMO TESTING)
- NON-TAILORED THERMO-STRUCTURES APPROACH LEADS TO ARTIFICIAL WEIGHT/ BALANCE PENALTIES
- IDLE MODE PENALTIES OUTWEIGH BENEFITS
- SYSTEM IS MORE SENSITIVE TO PAYLOAD GEOMETRY THAN WEIGHT
- DOMINANT SYSTEM SENSITIVITIES:
  - GO-AROUND DEFINITION
  - ORBITER ENGINE OUT APPROACH
  - CARRIER RETURN TO LAUNCH SITE
  - ORBITER INERT WEIGHT CONTINGENCY
  - BOOST ENGINE SPECIFIC IMPULSE

ILRVS-495F

RECOMMENDATIONS FOR FURTHER STUDY

It is recognized that a number of items have not been addressed to the extent necessary to draw firm conclusions and should therefore be pursued subsequent to this study. A listing of the key areas is shown on the facing page.

Internal support and heat-short penalties must be further defined for the integral tank concept.

Full scale structural element fabrication is necessary to realize a high confidence weight estimate.

The requirement for thermo-structures optimization and high-confidence boundary layer transition criteria are self-evident.

Ascent aero/thermodynamics testing is essential as this area is not amendable to rigorous analysis. The potential problems inherent in emergency staging in the atmosphere can only be addressed when the aerodynamics of the ascent configuration is known.

The articulation of airbreathing engines is highly unconventional and their operation in a possibly unfavorable flow field must be assessed.

The requirement for go-around capability and the degree of capability must be further examined because of the large potential penalties.

Interface requirements can be identified only when both elements are known. No space station definition was available for this study.

Launches from ETR cannot guarantee safe abort for all azimuths without unacceptable cruise penalties on the orbiter. A continental launch can offer such a guarantee, but other drawbacks such as new facilities, populace safety, and air traffic interference exist.

Without firm handling qualities criteria specifically derived for the large vehicles considered, no complete and realistic assessment of competing configurations can be made.

As noted in the preceeding chart, payload shape modifications may offer improved system performance.

RECOMMENDATIONS FOR FURTHER STUDY

- INTEGRAL TANK DESIGN (HL–10)
- FULL SCALE STRUCTURE ELEMENT FABRICATION
- THERMO-STRUCTURES OPTIMIZATION
- BOUNDARY LAYER TRANSITION CRITERIA
- ASCENT AERO/THERMODYNAMICS TESTING
- EMERGENCY STAGING DYNAMICS (ATMOSPHERE)
- A/B ENGINE DEPLOYMENT/OPERATION
- GO-AROUND REQUIREMENTS
- SHUTTLE/SPACE STATION INTERFACE
- ALL AZIMUTH ABORT CAPABILITY
- HANDLING QUALITIES CRITERIA – LARGE VEHICLES
- PAYLOAD SHAPE OPTIMIZATION

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